SCOUT

user's manual

THIS ISSUE SUPERSEDES ALL PREVIOUS ISSUES OF SCOUT USER'S MANUAL

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SCOUT USER'S MANUAL

Vought Corporation

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2.0 VEHICLE DESCRIPTION

The standard Scout* launch vehicle is a solid propellant four stage booster system providing an efficient means of boosting a spacecraft on a planned trajectory. The Scout vehicle is capable of orbit, probe, and re-entry earth missions with demonstrated high reliability. A standard fifth stage is available for highly elliptical and solar orbit missions.

Vought Corporation is the prime contractor to the National Aeronautics and Space Administration for the Scout launch vehicle and as such is responsible for the launch vehicle system.

The basic Scout vehicle is made up of the following major assemblies: rocket motors, base A, structural transition sections and a heatshield. The inboard profile of the standard Scout vehicle is shown in Figure I-1.

Each transition section is divided into lower and upper portions at the stage separation plane. A frangible diaphragm separation system is used in transition sections B and C. A spring ejection system is used in transition section D and the section F for the optional fifth stage.

The Scout vehicle is equiped with a preprogrammed guidance system which commands the separation of each expended stage on a timed sequence. The payload is protected from the high temperatures during ascent by a two piece heatshield which is ejected just prior to third stage ignition.

Normally, the yaw and roll axes are maintained at the launch reference while the pitch axis is programmed through a pre-selected angle corresponding to the desired vehicle trajectory. However, a yaw torquing capability is incorporated which also provides programming of the yaw axis should this maneuver be required.

A proportional control system is contained in the base A featuring combination of jet vanes and aerodynamic tip control surfaces, operated by hydraulic servo actuators, and is used to control the vehicle throughout the entire first stage period.

FIGURES

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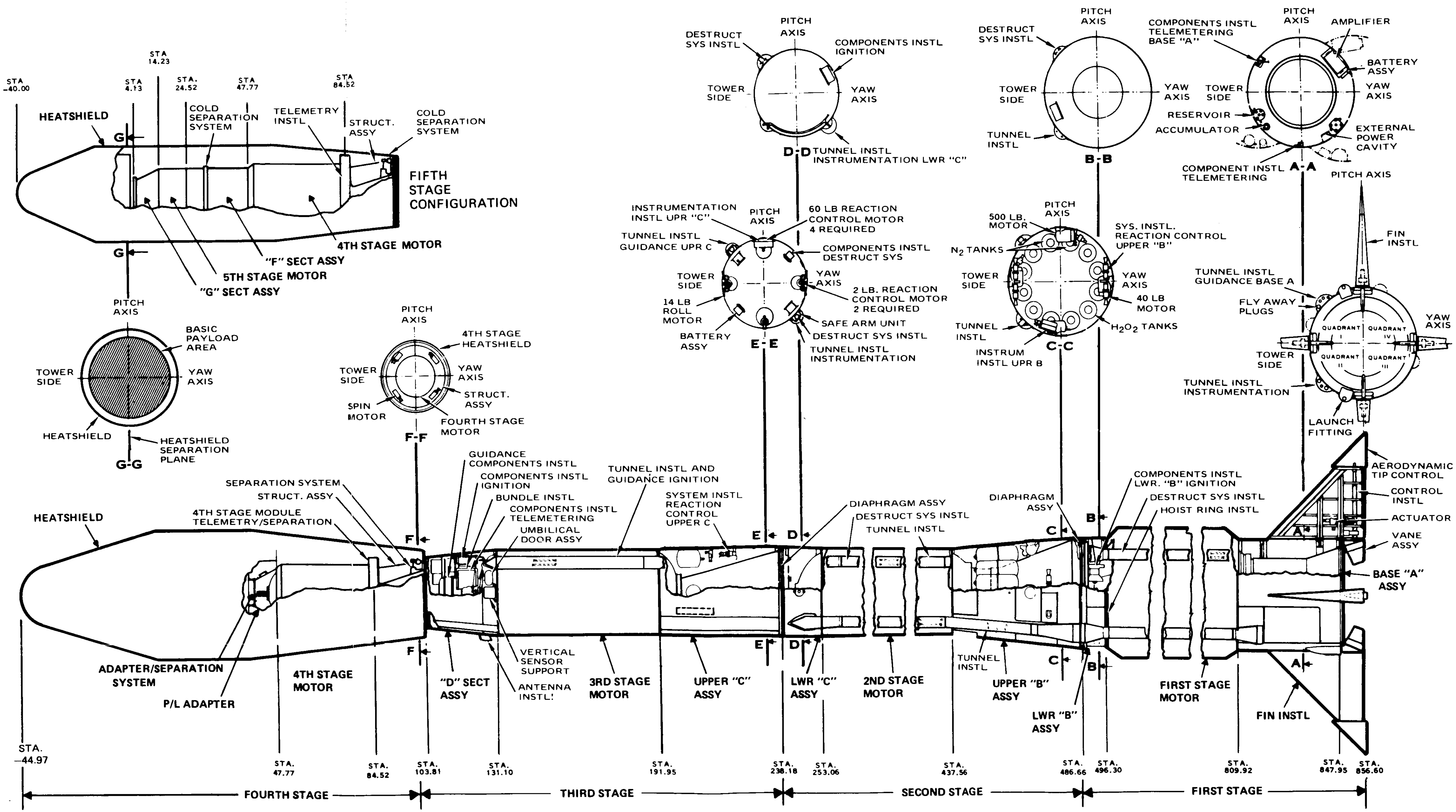


FIGURE I–1 SCOUT INBOARD PROFILE

1-4-a

15-b 15-a

15-a

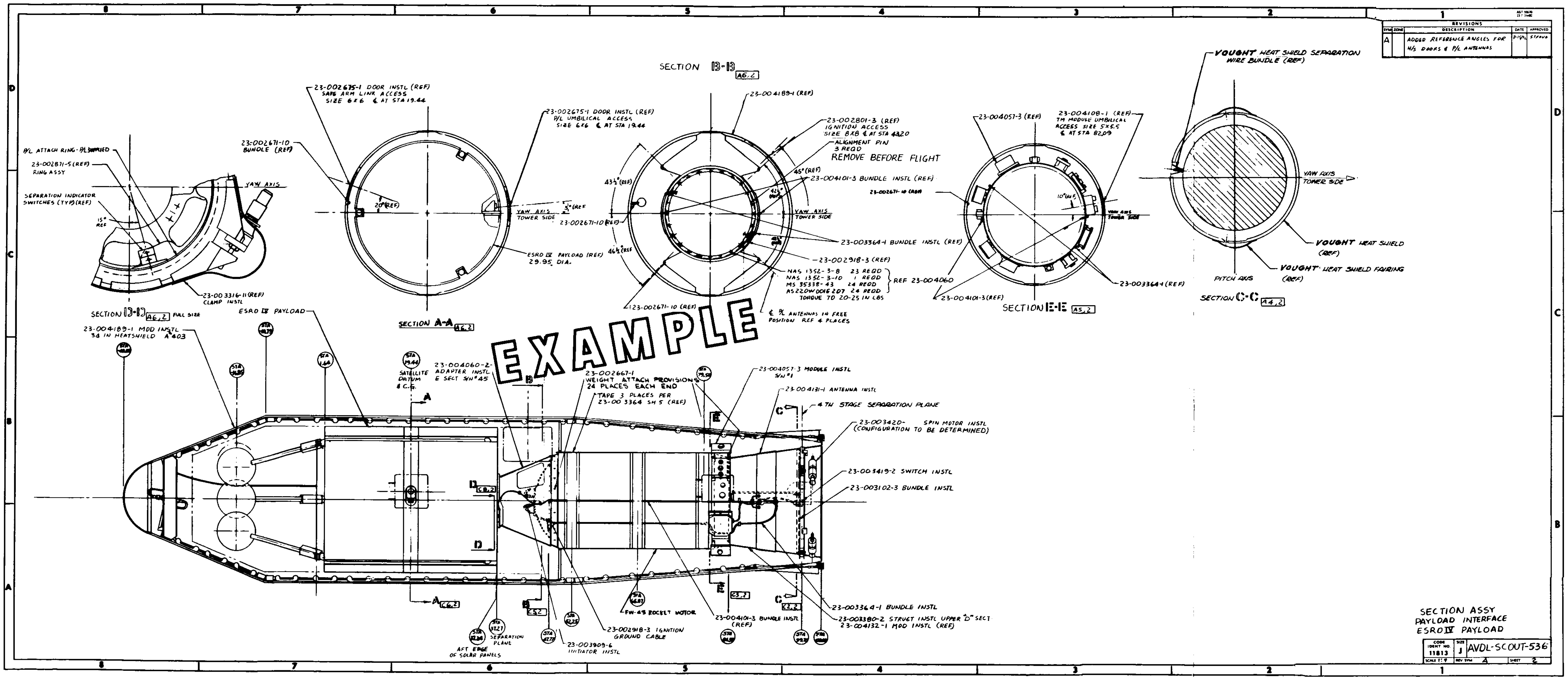
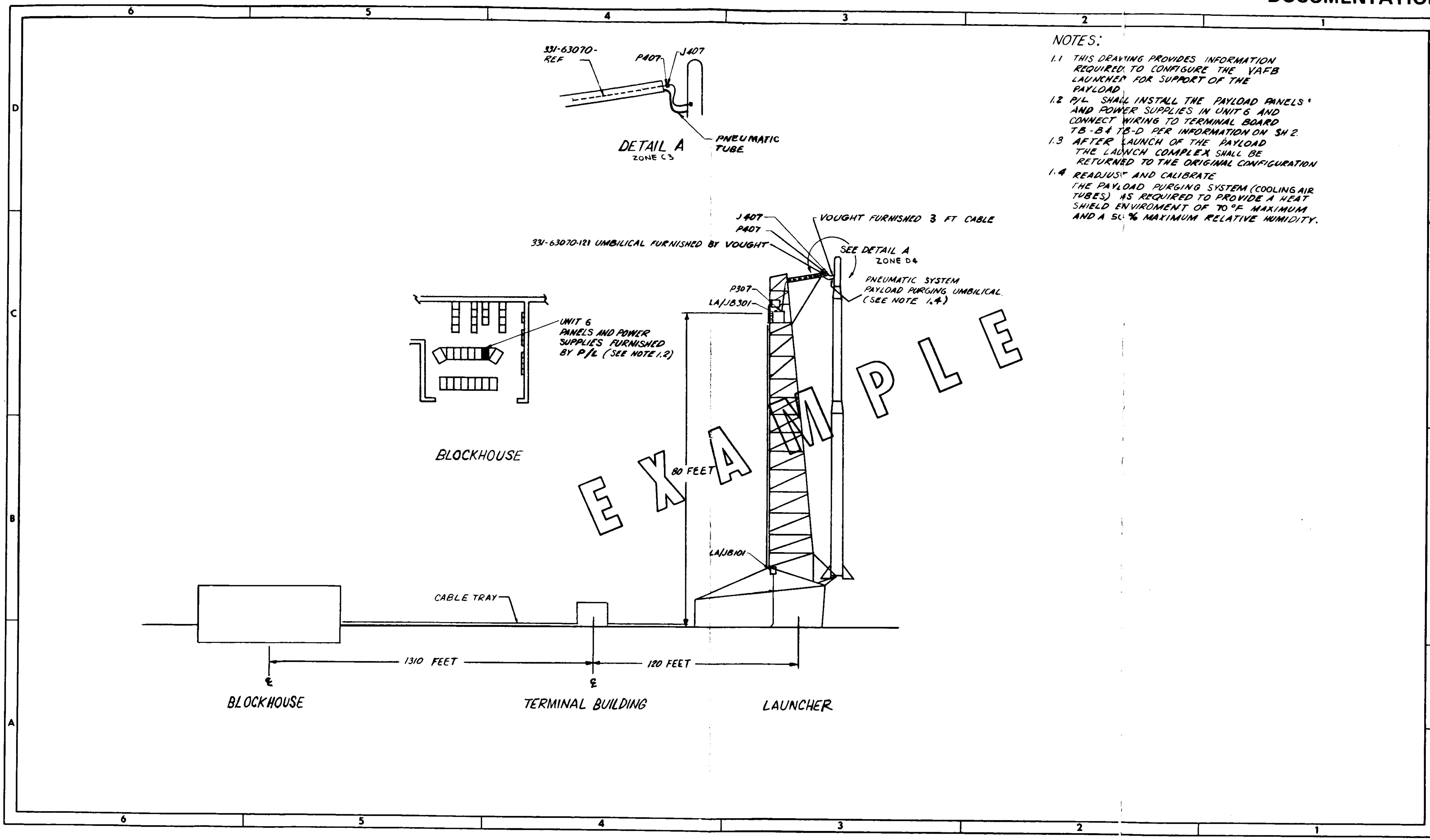


FIGURE III–3 TYPICAL PAYLOAD INTERFACE DRAWING – VEHICLE

71-a

71-B 71-a



72-a

FIGURE III–4 TYPICAL PAYLOAD INTERFACE DRAWING – GROUND SUPPORT EQUIPMENT (SHEET 1 OF 2)

72-B

72-a

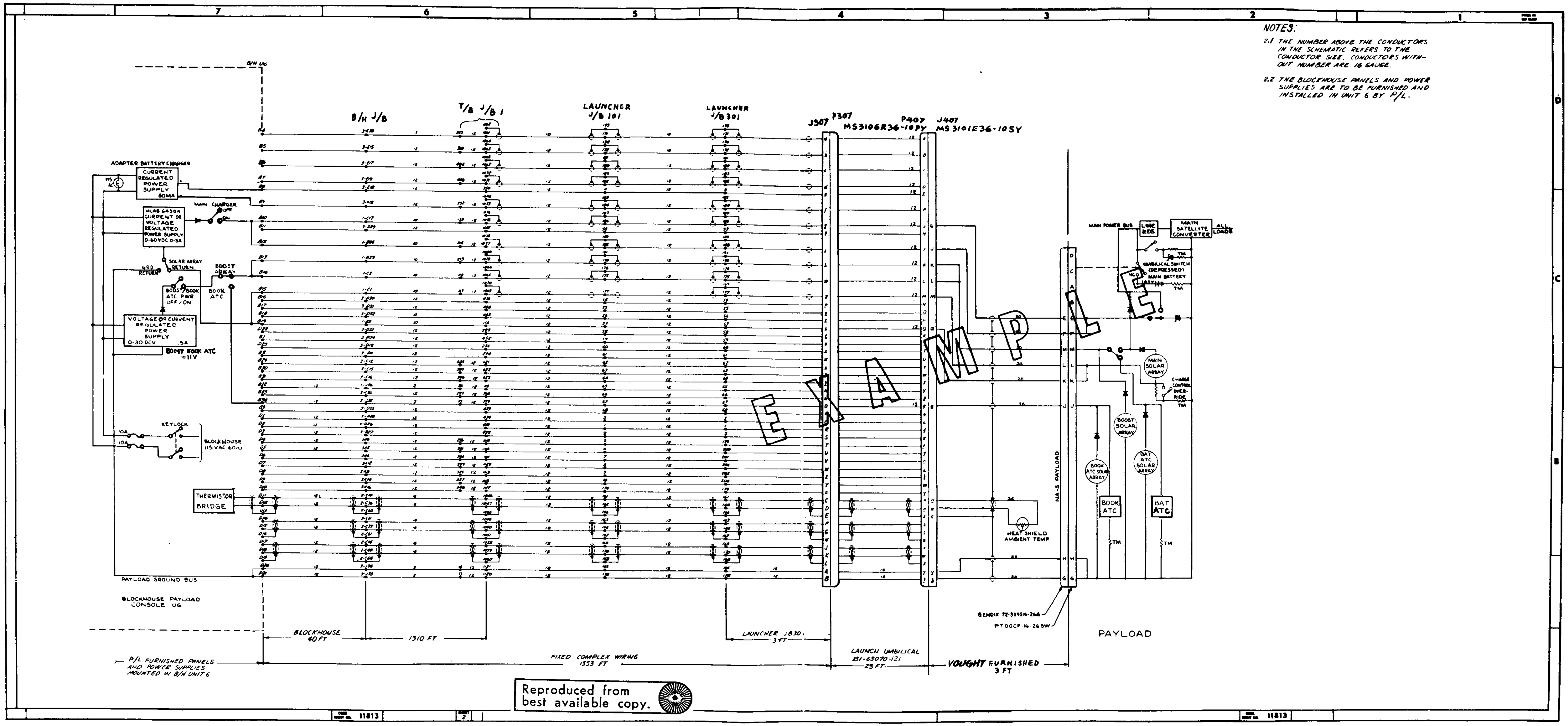
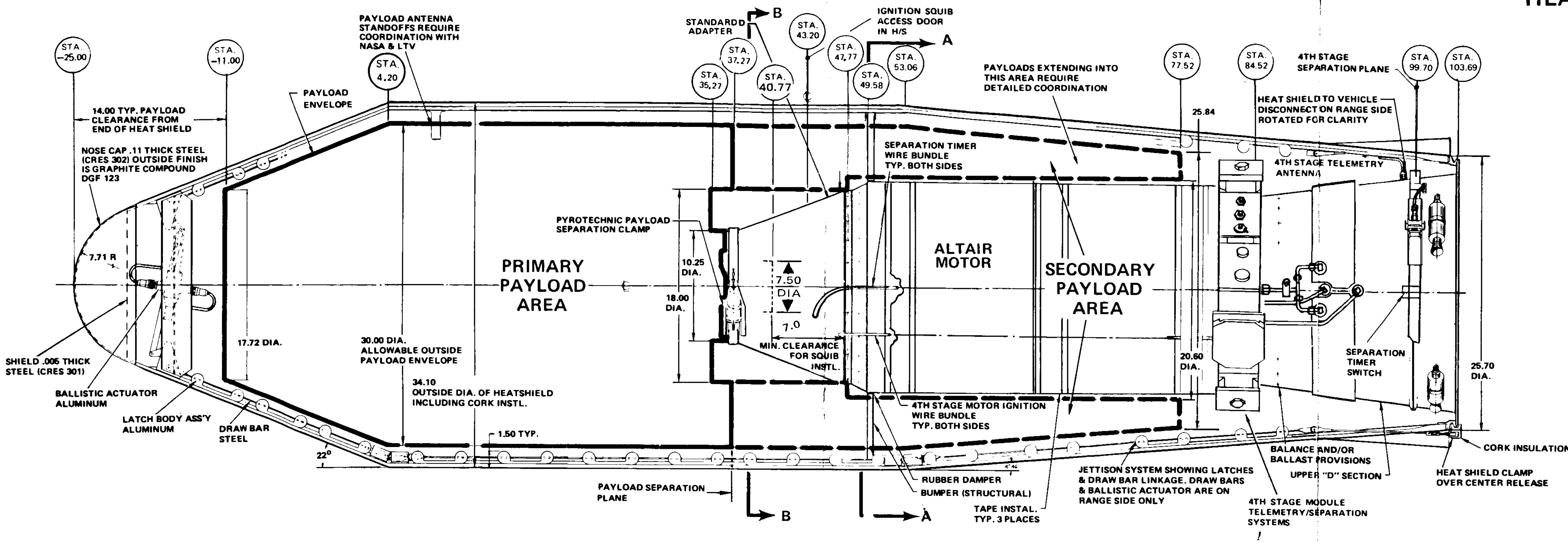


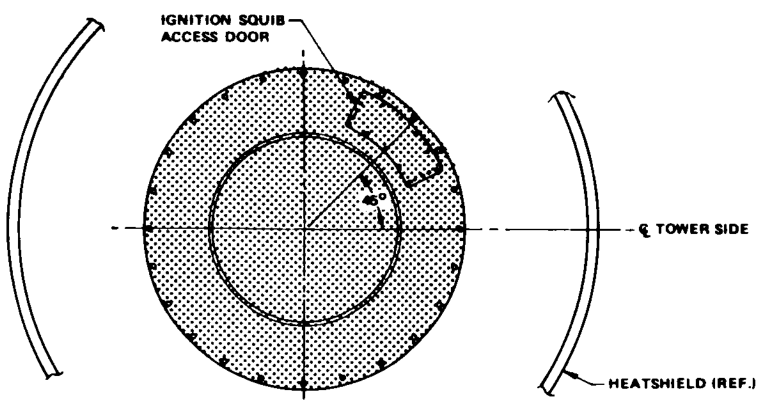
FIGURE III–4 TYPICAL PAYLOAD INTERFACE DRAWING – GROUND SUPPORT EQUIPMENT (SHEET 2 OF 2)

73-a

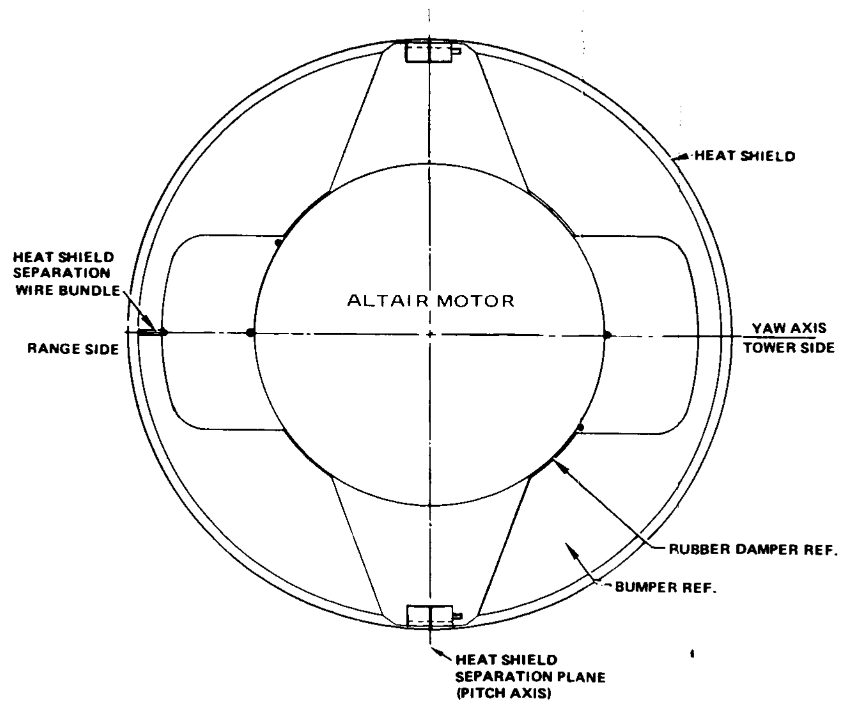
73-a

73-B





SECTION B–B



SECTION A–A
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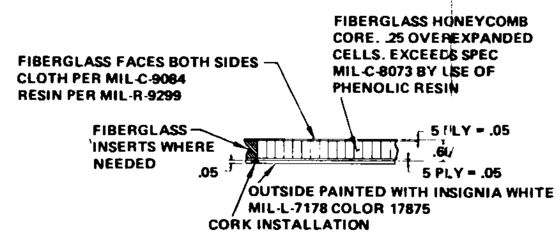

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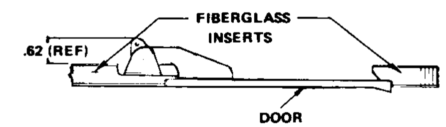


TYP DOOR SECTION
SPRING LOADED

FIGURE V–19 SCOUT FOURTH STAGE PAYLOAD ENVELOPE, 34 INCH DIAMETER (.86 METER DIA.) –25 NOSE STATION

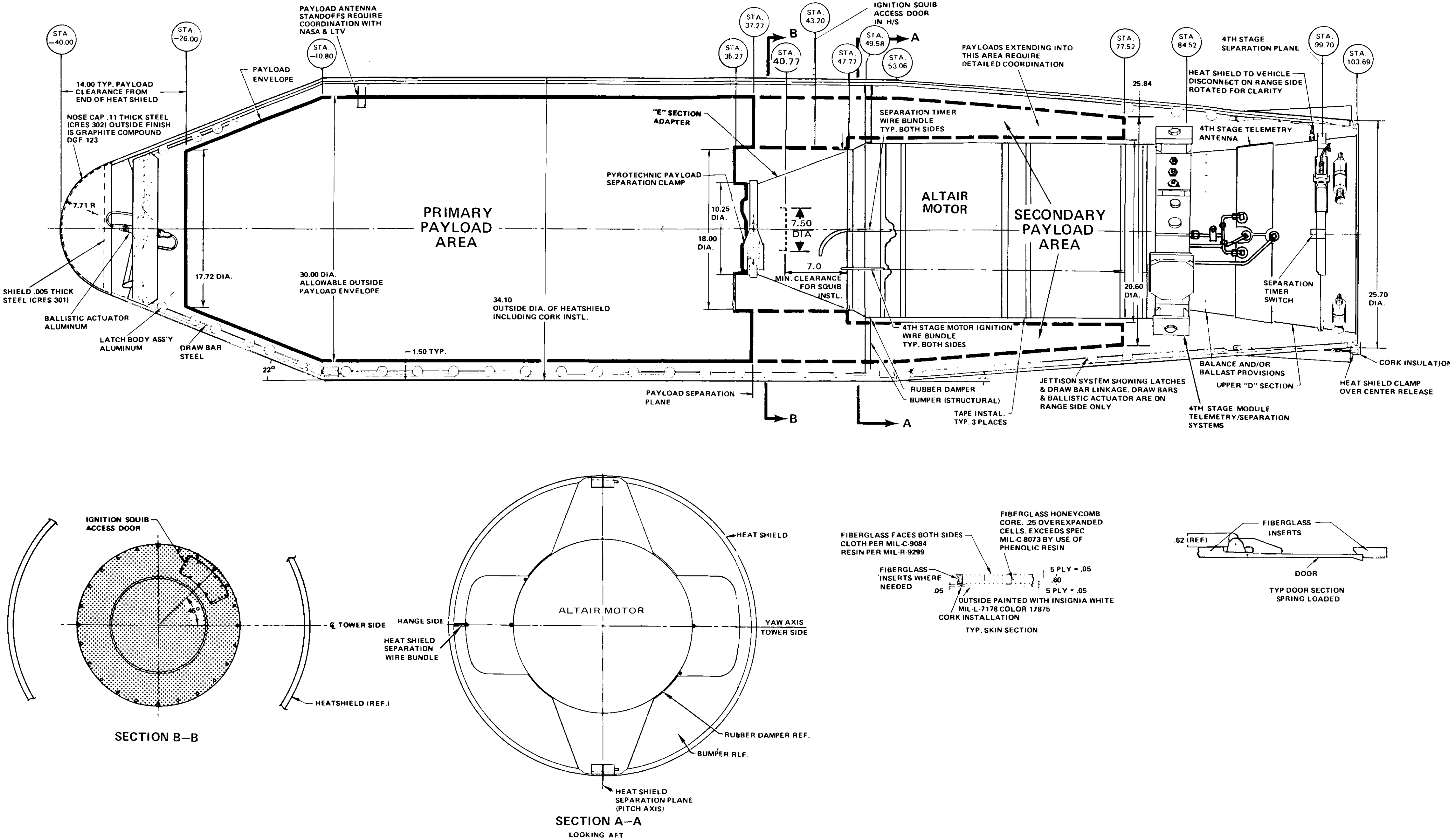
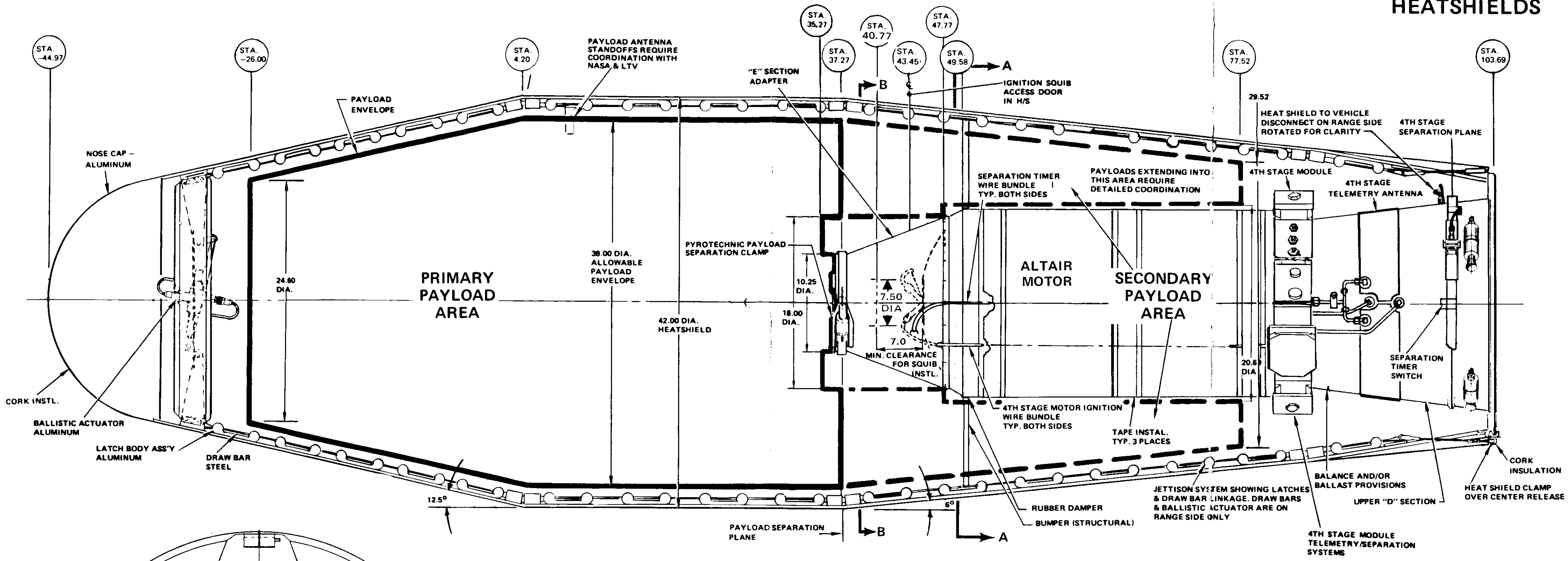


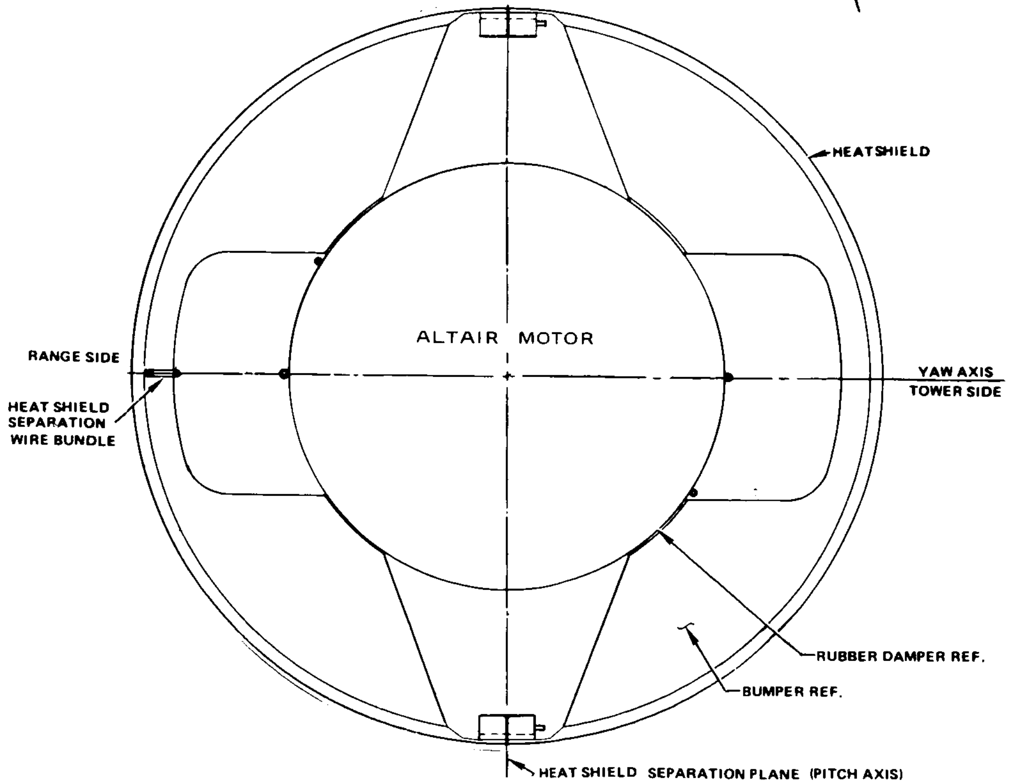
FIGURE V—20 SCOUT FOURTH STAGE PAYLOAD ENVELOPE, 34 INCH DIAMETER (.86 METER DIA.) — 40 NOSE STATION

172-a

172-a

172-B





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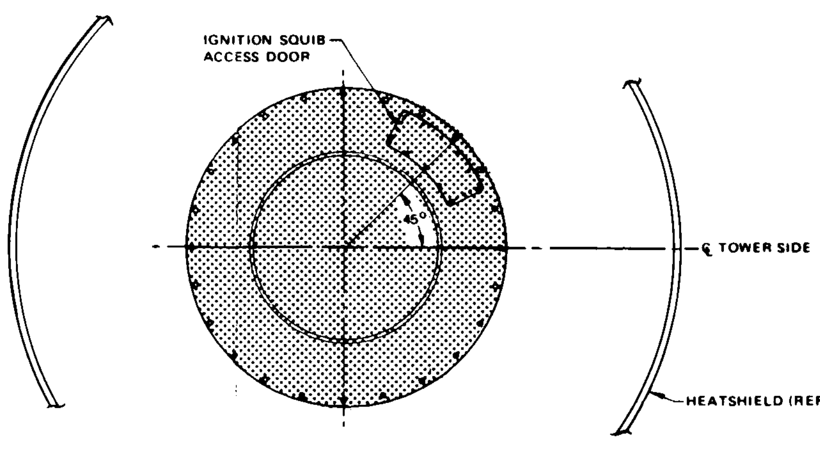
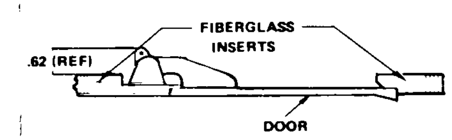
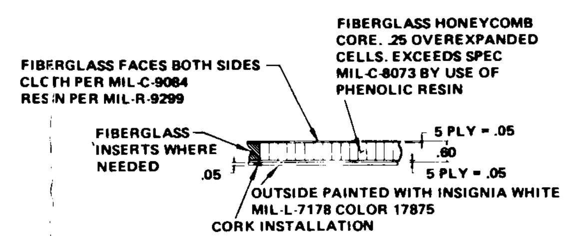
LOOKING AFT



SECTION B–B



TYP DOOR SECTION
SPRING LOADED

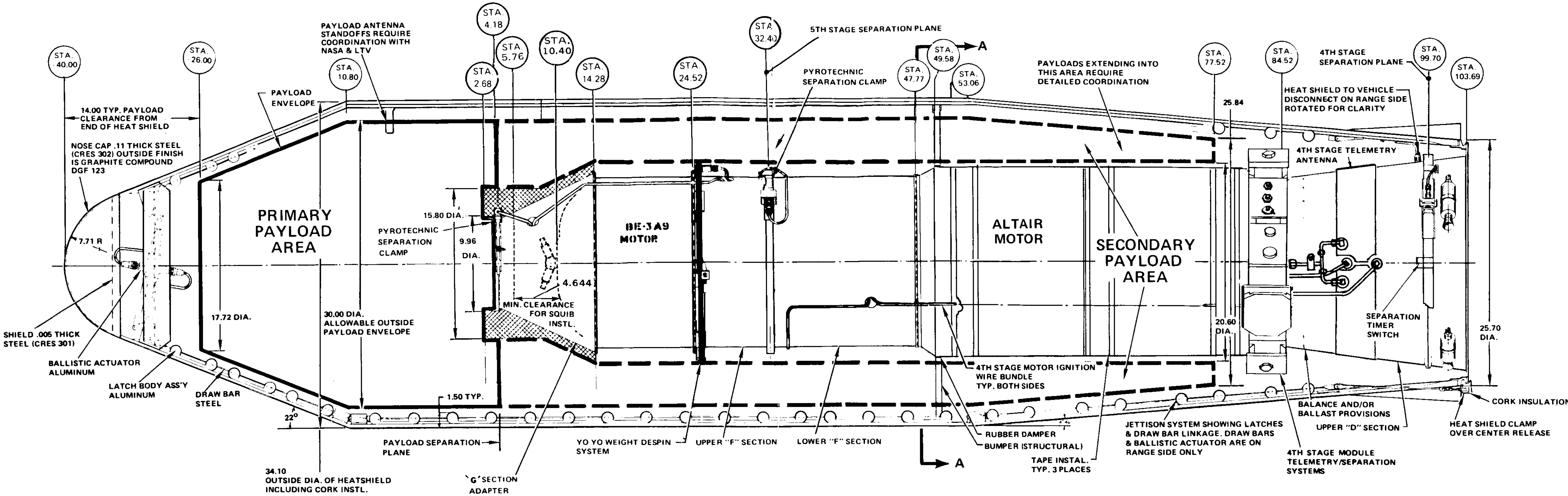


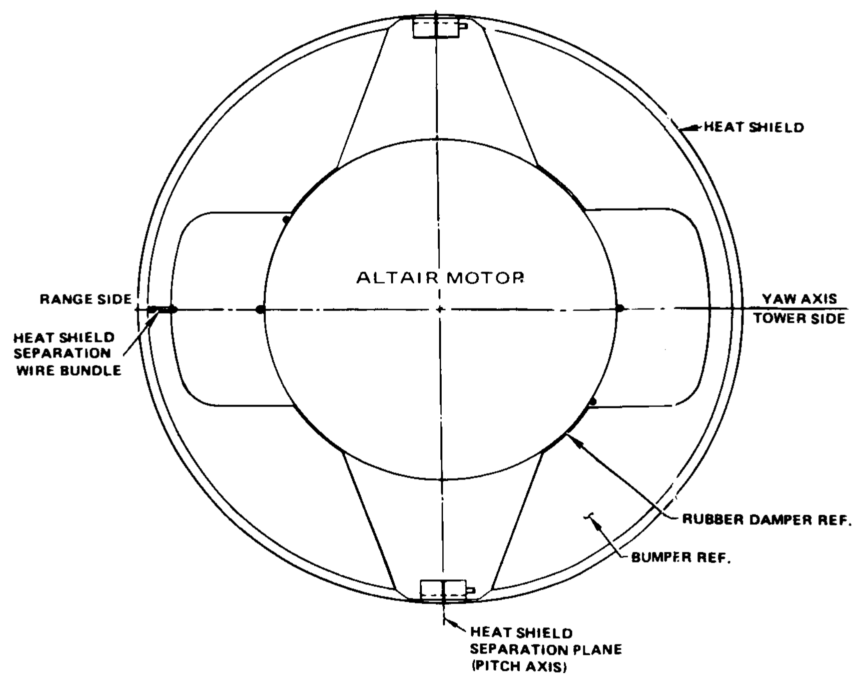
TYP. SKIN SECTION

FIGURE V–21 SCOUT FOURTH STAGE PAYLOAD ENVELOPE, 42 INCH DIAMETER (1.07 METER DIA.) –45 NOSE STATION

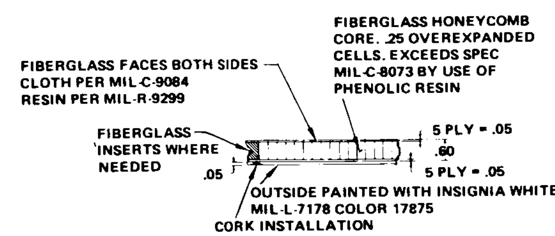
173-a

173-a
173-B

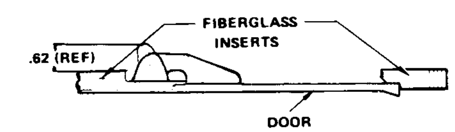




SECTION A
LOOKING AFT



TYP. SKIN SECTION



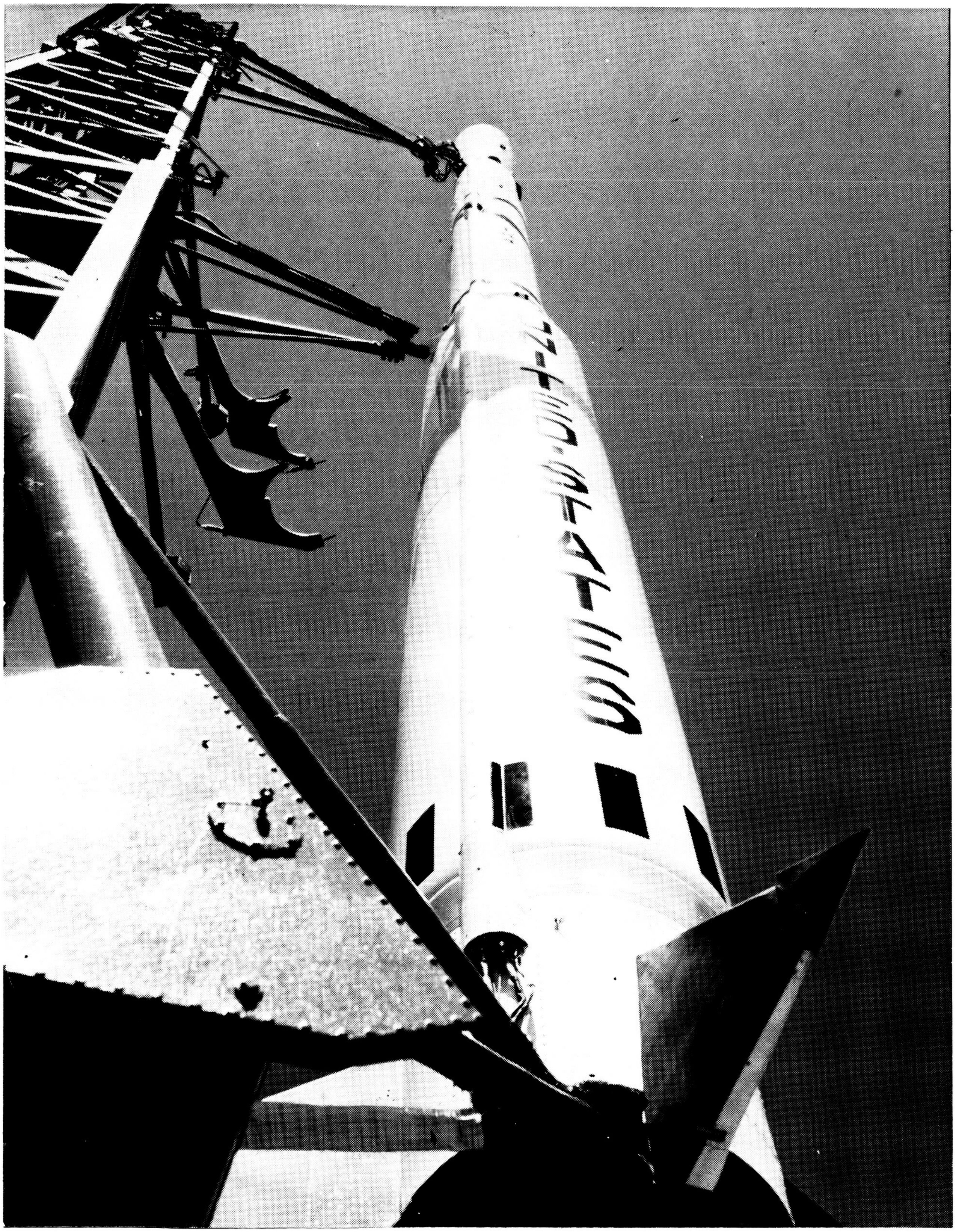
TYP DOOR SECTION SPRING LOADED

FIGURE V–22 SCOUT FIFTH STAGE PAYLOAD ENVELOPE, 34 INCH DIAMETER (.86 METER DIA.) –40 NOSE STATION

174-a

174-a

174-b



SCOUT STANDARD LAUNCH VEHICLE

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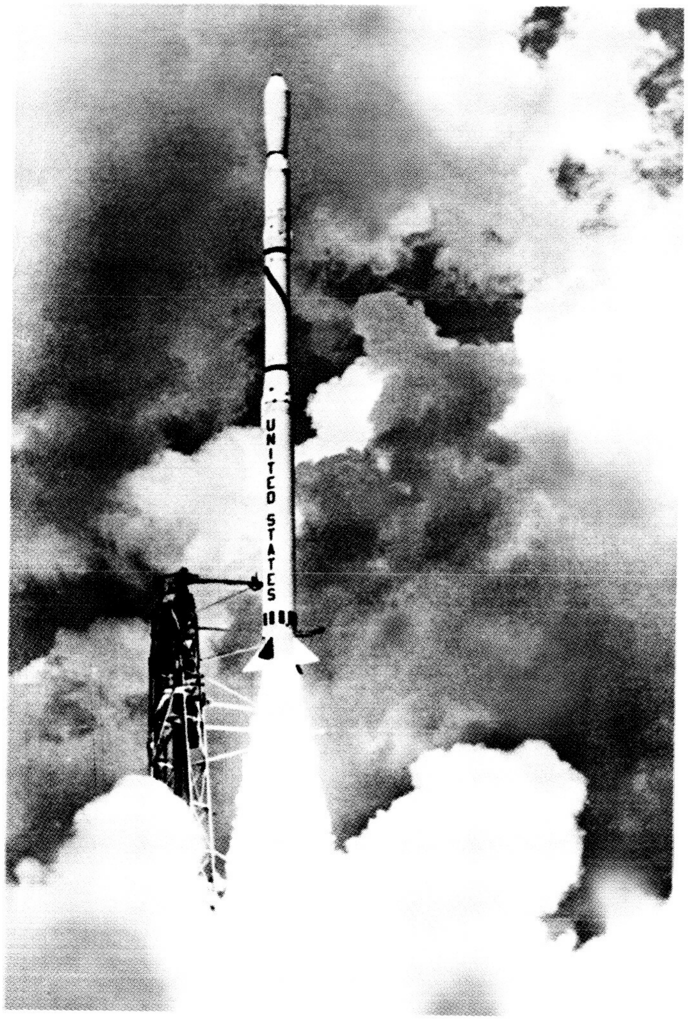
chapter I
INTRODUCTION



SCOUT
user's manual

1.0 INTRODUCTION

The purpose of this manual is to assist the payload design engineer in integrating his payload to the Scout vehicle. The information presented herein is intended to:

- Acquaint the Scout User with the Standard Scout Launch Vehicle and its performance capabilities.
- Identify the management organization administering Scout programs and the required documentation.
- Describe the Scout launch sites, support facilities and operations.
- Specify the payload interface design criteria including vehicle, flight environment and operational constraints.

Data for every Scout vehicle configuration and option combination that is now available or is being developed is not included in this manual. The data presented includes only the more widely used configurations. Other data can be furnished as required.

Note

The Design Data and Performance Curves in this manual are intended for preliminary planning and feasibility studies only. Final design information shall be confirmed through the Mission Working Group.

Second and third stage control systems are based on the same concept of operation as the first stage but differ in the method used to generate the control force. The control forces for these two stages are provided by hydrogen peroxide reaction motors which are operated as an "on-off" system.

The unguided fourth stage (with optional fifth stage) is spin stabilized, prior to separation from the third stage, at the reference attitude established at separation.

2.1 TYPICAL MISSION PROFILE

A pictorial description of a typical orbital four stage Scout mission profile is shown in Figure I–2.

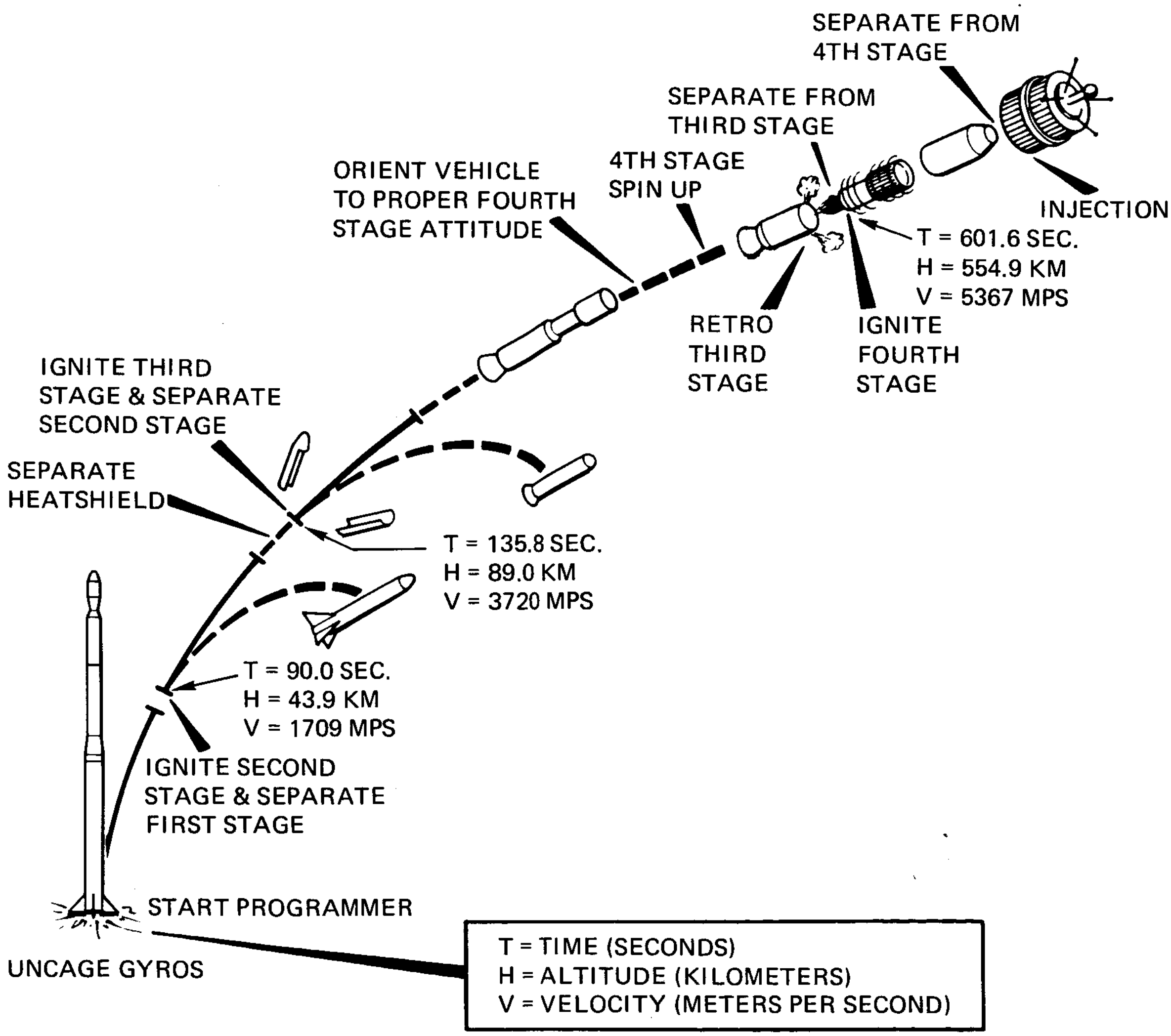


FIGURE I–2 SCOUT MISSION PROFILE

2.2 VEHICLE CONFIGURATIONS

Since the launch of the first Scout vehicle in 1960, there has been a continued program of improvements in Scout performance capabilities resulting in several vehicle configurations. The vehicle configurations are identified by letters to designate the propulsion system rocket motor combinations. Four and five stage configurations have been established and are summarized in Table I-1.

TABLE I-1 STANDARD VEHICLE CONFIGURATIONS

Config.	1st Stg. Mtr.	2nd Stg. Mtr.	3rd Stg. Mtr.	4th Stg. Mtr.	5th Stg. Mtr.
FIVE STAGE	Algol IIIA	Castor IIA	Antares IIIA	Altair IIIA	Alcyone IA (optional)
FOUR STAGE	Algol IIIA	Castor IIA	Antares IIIA	Altair IIIA	

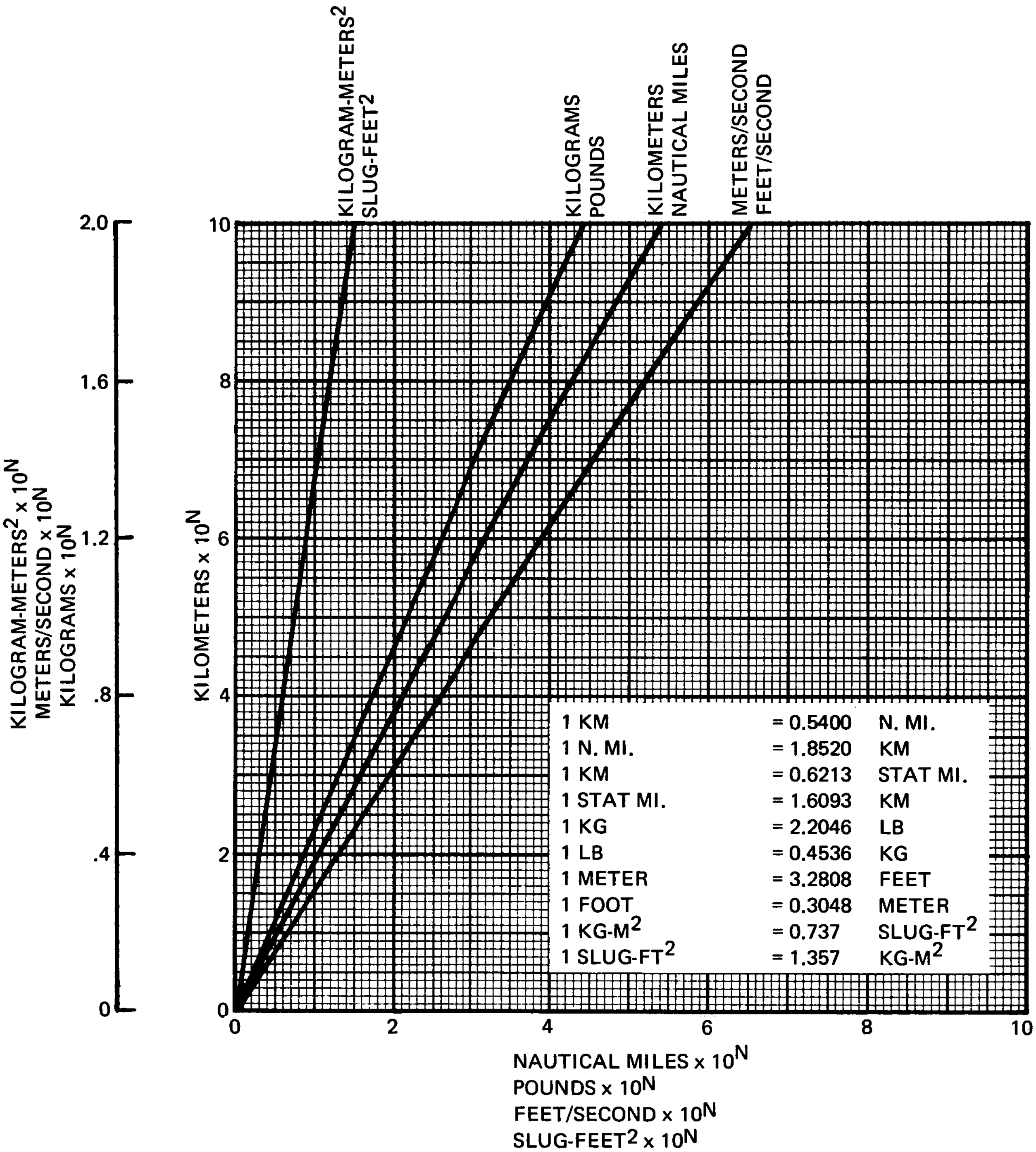
The data presented throughout this manual are predominately for the four stage configuration with the five stage configuration covered only selectively.

Various combinations of standard heatshields and payload adapter sections are available for use with each vehicle configuration. The standard options are summarized in Table I-2. The standard adapter sections listed were designed to provide a separation system when required. The Payload Agency may choose, as an additional option, to furnish its own payload to vehicle adapter section and/or separation system. Use of a Payload Agency furnished adapter/separation system will require written prior approval of the Mission Working Group.

TABLE I-2 VEHICLE STANDARD OPTIONS

CONFIGURATIONS	ADAPTER SECTIONS WITH OR WITHOUT SEPARATION SYSTEMS	HEATSHIELDS
FIVE STAGE	"G"	34, −40; 42, −45
FOUR STAGE	"EG", SERIES 200 "E" SERIES 25 "E"	34, −25; 34, −40; 42, −45

I'l



1 KM	= 0.5400	N. MI.
1 N. MI.	= 1.8520	KM
1 KM	= 0.6213	STAT MI.
1 STAT MI.	= 1.6093	KM
1 KG	= 2.2046	LB
1 LB	= 0.4536	KG
1 METER	= 3.2808	FEET
1 FOOT	= 0.3048	METER
1 KG-M^2	= 0.737	SLUG-FT^2
1 SLUG-FT^2	= 1.357	KG-M^2

ENGLISH-METRIC CONVERSION CHART

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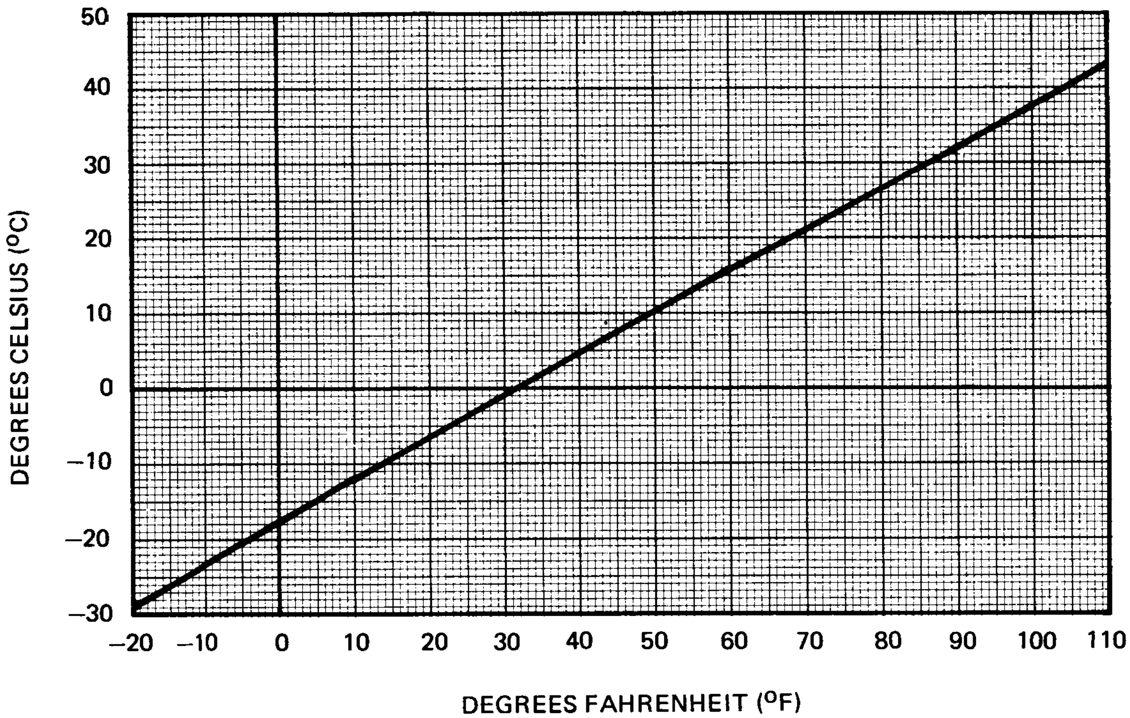
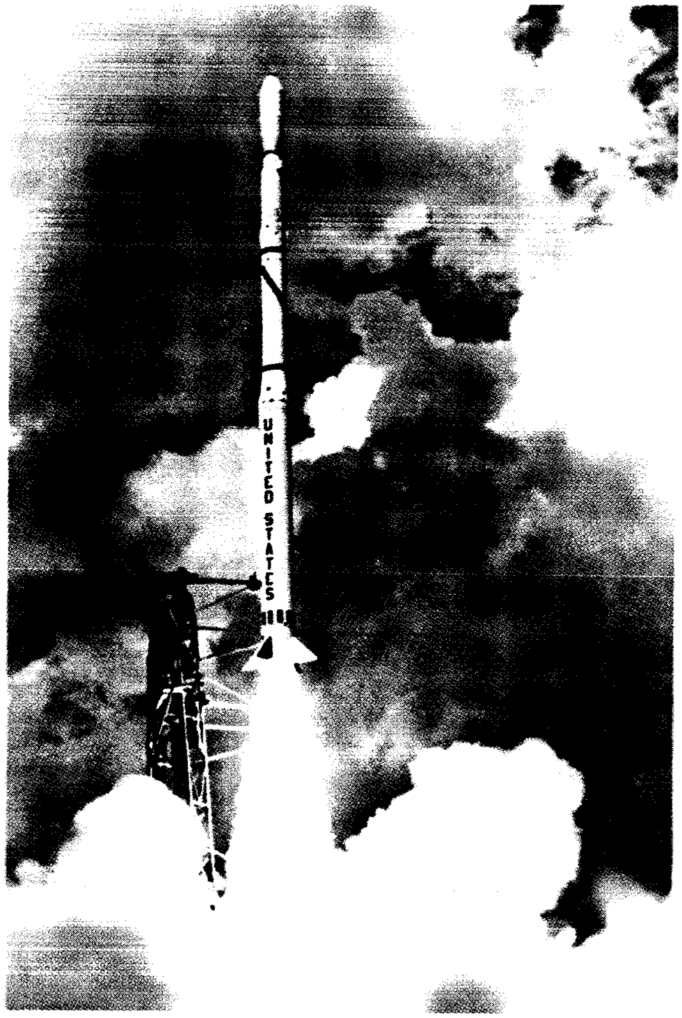


TEMPERATURE CONVERSION

chapter II

PERFORMANCE CAPABILITY



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1.0 PERFORMANCE AND FLIGHT PLANNING

This chapter provides information on Scout vehicle performance capabilities. The vehicle performance data includes orbital, earth probe, re-entry and solar orbit missions; payload-inclination relationships; orbital accuracy; and circular orbit lifetimes. The Scout orbital launches to date are listed and the orbital accuracy is summarized.

Since performance capability for any given mission is dependent on mission requirements and range safety considerations, flight planning is briefly discussed with respect to pre-flight trajectory analysis, description of mission and selection of launch site.

1.1 FLIGHT PLANNING

A major objective of Flight Planning is to determine the optimum flight trajectory that satisfies the mission requirements and is consistent with vehicle capabilities and Range Safety considerations. The trajectory is important to the Payload Agency as it accurately defines the vehicle/payload space-time position. It is important to range safety in order that the safety aspects of the mission can be evaluated. Trajectory data are determined by a Pre-Flight Trajectory Analysis and are documented by the Trajectory Data Package for each mission.

In some instances of specialized mission requirements, Preliminary Trajectory Studies or Mission Feasibility Studies are required to assist the user of Scout to define a set of final mission requirements.

A Pre-Flight Trajectory Analysis requires inputs that include, but are not limited to, payload definition, mission requirements, launch site selection, payload environmental restrictions and rocket motor performance predictions. These inputs must be provided by the applicable agencies and submitted through the Mission Working Group as prescribed in the Documentation Requirement Section of Chapter III.

The final pre-flight predicted trajectory data is required by the ranges thirty days prior to launch. To meet this requirement, the mission description and configurational data are required at least sixty days prior to launch. Preliminary preflight analysis will normally begin nine months before launch.

1.1.1 Mission Description

The mission description required to initiate either a preliminary analysis or to accomplish the final pre-flight analysis is dependent upon the type of mission to be flown. There are four basic mission types to which the Scout vehicle is suited. These are earth probe, high velocity re-entry, solar orbit and near earth orbit missions.

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Table II-1 summarizes for each mission type the general requirements for mission description necessary to perform a pre-flight trajectory analysis. Mission requirements should be defined and documented in the Payload Description Document.

The Payload Agency should specify the base upon which orbital altitudes are to be defined. If this base is not specified, the altitudes will be based on a equatorial earth radius. Scout designs the ascent boost trajectory so that injection coincides with the perigee. The practice will always be followed in the design of pre-flight trajectories unless a particular argument of perigee is desired. In this event, the pre-flight trajectory will be designed to conform with this requirement, but the Scout user should be aware that this maneuver entails a performance loss and affects the vehicle accuracy.

TABLE II–1 MISSION DESCRIPTIONS REQUIRED FOR PRE-FLIGHT ANALYSIS

	NORMAL REQUIREMENTS	ADDITIONAL REQUIREMENTS
ORBITAL MISSIONS	INCLINATION AND TOLERANCE APOGEE ALTITUDE (1) AND TOLERANCE PERIGEE ALTITUDE (1) AND TOLERANCE PAYLOAD WEIGHT C.G. AND INERTIA DESIRED SPIN RATE	ORBIT LIFE TIME REQUIREMENT HEAT SHIELD EJECTION ALTITUDE COEFFICIENT OF DRAG AND REFERENCE AREA ORBITAL PRECESSION REQUIREMENT LAUNCH WINDOW AND DATE
RE-ENTRY MISSIONS	ENTRY ALTITUDE (LOCAL) & TOLERANCE ENTRY ANGLE (LOCAL) & TOLERANCE ANGLE OF ATTACK AND TOLERANCE EARTH RELATIVE VELOCITY AND TOLERANCE PAYLOAD WEIGHT, C.G., AND INERTIA DESIRED SPIN RATE	RANGE AND TOLERANCE GEOGRAPHIC POSITION OF ENTRY AND TOLERANCE DATA ACQUISITION VIEWING CONSTRAINTS
PROBE MISSIONS	PEAK ALTITUDE (LOCAL) OR TIME IN WEIGHTLESS CONDITION PAYLOAD WEIGHT, C.G., AND INERTIA DESIRED SPIN RATE	PEAK ALTITUDE POSITIONING FOURTH STAGE IMPACT AREA FOR POSSIBLE RECOVERY DATA ACQUISITION VIEWING CONSTRAINTS

NOTES:

(1) SPECIFY IF ALTITUDES ARE TO BE BASED ON OTHER THAN MEAN EQUATORIAL RADIUS.

PERFORMANCE AND FLIGHT PLANNING

1.1.2 Range Safety Analysis

Another aspect of flight planning is the choice of launch site. The major factors include the type of mission, specific mission requirements, and range safety considerations.

A Range Safety Analysis is performed by the applicable range for most Scout missions. Certain trajectory data are required by the ranges to be used in support of their Range Safety Analysis. The Trajectory Data Package prepared by Vought Corporation is utilized by the ranges to perform this analysis. Stage impact dispersion data from the Pre-Flight Brochure is also utilized.

Range safety inputs are submitted through the Mission Working Group as prescribed in the Documentation Requirements Section of Chapter III.

1.1.3 Trajectory Data Package

In the support of each Scout launch, a package of trajectory data is provided to all pertinent agencies. This package contains the actual computer trajectory listings in the support of the launch. These listings will vary depending upon the mission type and range to be used. The following list of items are generally included in the trajectory package and those items which are keyed to either mission type or range used are noted.

Items included in Trajectory Data Package:

1. Nominal boost trajectory to fourth stage burnout.
2. Expended stage impact trajectories for first three stages.
3. Impact trajectory for expended fourth stage and payload (probe and re-entry missions only).
4. Radar tracking look-angle data (Wallops Flight Center launches only).
5. Instantaneous impact points for first three stages.
6. Ephemeris of predicted orbit.
7. Magnetic tapes.

1.1.4 Pre-Flight Brochure

Approximately thirty days prior to the scheduled launch of a Scout vehicle, the Pre-Flight Brochure is published which contains technical information pertinent to the subject vehicle. The data presented included such items as the vehicle commanded pitch program, trajectory parameters at event times and predicted impact dispersion areas for the various stages (and payload when applicable). The brochure provides a ready source of reference for the launch preparation. This brochure is presented to the Payload Agency as a matter of course and need not be requested.

2.0 PERFORMANCE DATA

As a result of the continuing growth program of the Scout vehicle, there are now available to the Scout User several vehicle configurations to meet the varying mission requirements. These configurations are described in Chapter I. It is not practical to present the complete spectrum of performance of all possible vehicle configurations and heatshield combinations for the various types of missions and launch sites. Only the performance data for the more widely used configurations are provided. Performance data for configurations not included can be made available as required.

Scout performance capabilities are generally presented for configurations with the 1.07 meter heatshield. Where performance capability is given for Scout configurations using the 0.86 meter diameter heatshield the performance capability for Scout configurations using the larger 1.07 meter diameter heatshield can be obtained by decreasing the values shown by five percent. Conversely, to obtain performance capabilities for configurations using the smaller heatshield, increase the values shown for the large heatshield by five percent.

An English-Metric conversion chart has been provided at the end of Chapter I.

2.1 ROCKET MOTOR PERFORMANCE CHARACTERISTICS

The propulsion system of the Scout vehicle is composed of solid propellant rocket motors. Table II-2 lists the motors used and their performance characteristics.

TABLE II–2 SCOUT ROCKET MOTOR PERFORMANCE CHARACTERISTICS

STAGE	MOTOR DESIGNATION	TOTAL IMPULSE N–S VACUUM	AVG. WEB THRUST N VACUUM	WEB BURN TIME SEC	TOTAL WEIGHT Kg MASS
1	ALGOL IIIA	32,385,209	467,164	56.11	14,215
2	CASTOR IIA	10,254,867	284,316	35.32	4,433
3	ANTARES IIIA	3,736,200	83,100	43.96	1,394
4	ALTAIR IIIA	772,400	25,390	29.30	301.0
5 (OPTIONAL)	ALCYONE IA	236,024	27,492	8.42	98.2

NEWTON SECONDS X.225 = POUND SECONDS

2.2 PAYLOAD WEIGHT DEFINITION

It should be noted that all vehicle performance curves provided in this manual reflect total vehicle capability. In order to obtain spacecraft weight capability, such items as adapter/separation system (vehicle or payload furnished), separation system electrical components and any spacecraft peculiar items such as vehicle mounted solar paddle supports, must be subtracted from the total payload weight defined by the performance curves. The components included as payload weight are pictorially identified in Figure II-1. A component weights breakdown is included with the description of each component in the chapter on Design Parameters.

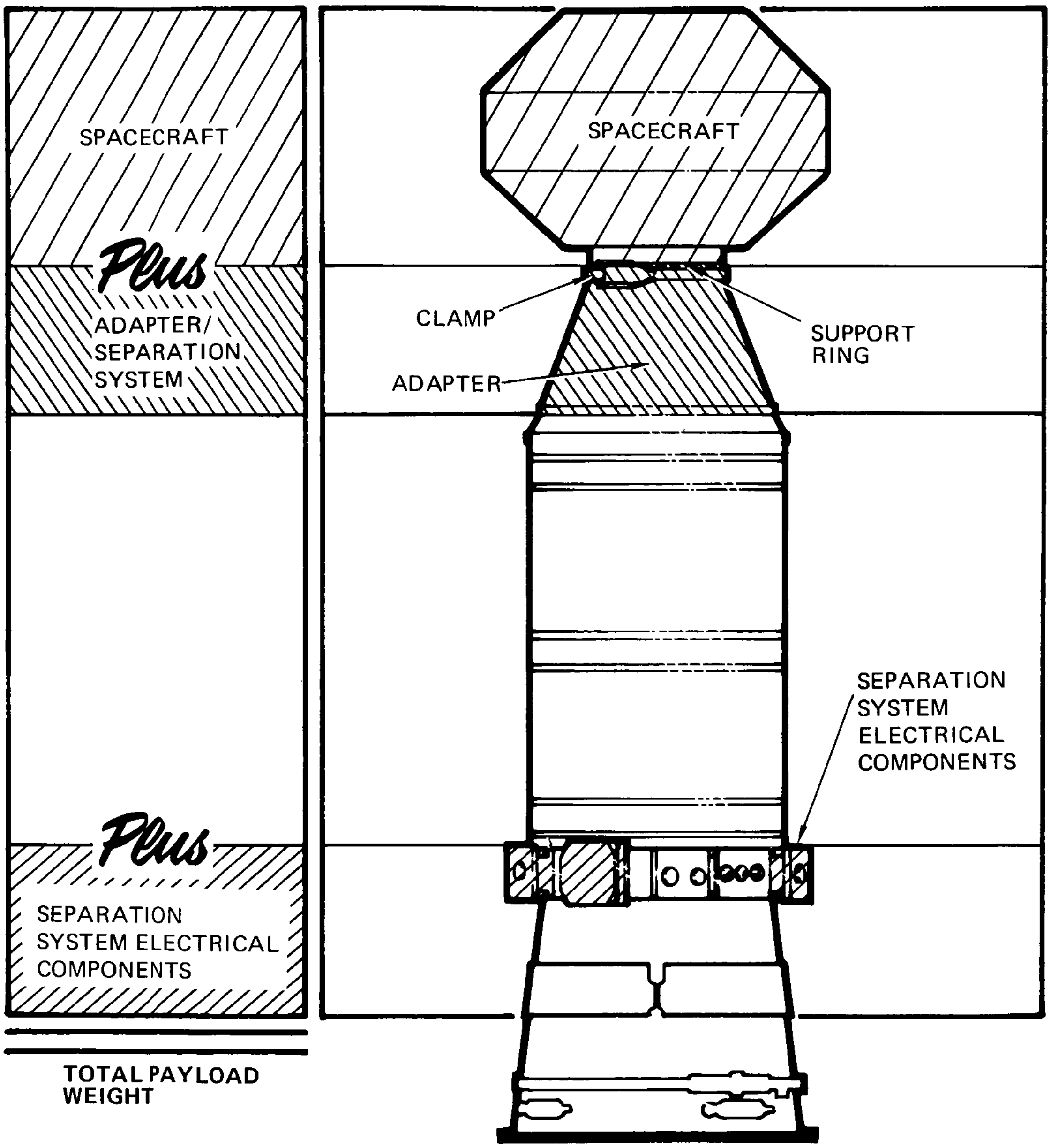

FIGURE II–1 PAYLOAD WEIGHT DEFINITION

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3.0 ORBITAL MISSION PERFORMANCE

The majority of Scout launch requirements have been aimed at achieving near earth orbits. Because of the strong emphasis in the past and in the forseeable future on orbit missions, the bulk of the performance data presented in this handbook will deal with this aspect of Scout capability.

For orbit missions the Scout vehicle utilizes the first three stages to boost the fourth stage plus spacecraft to the desired injection altitude. The fourth stage is then used as the injection stage to accelerate the spacecraft to the desired orbital velocity. The injection altitude is controlled by a predetermined pitch rate program which commands the guidance and control system to steer the vehicle along the desired trajectory. Orbit inclination is determined by the launch azimuth and any heading angle change obtained by a predetermined yaw rate program.

The second stage is not ignited until the dynamic pressure reduces to the level which insures adequate second stage controllability. A short coast phase following first stage burnout is sometimes necessary to obtain the proper dynamic pressure for second stage ignition. When the second stage is ignited, simultaneous separation from the first stage occurs.

A coast period of five seconds minimum follows second stage burnout during which the spacecraft heatshield is ejected. For adequate third stage controllability, the third stage is not ignited until dynamic pressure reduces to an acceptable level, thus extending second stage coast time beyond the minimum of five seconds. Third stage ignition and second stage separation then occur simultaneously. Normally, during the first three boost phases, the vehicle pitch rate program is designed to command the vehicle to fly a gravity turn (optimum) trajectory. For some missions, however, a non-optimum (shaped) trajectory is flown to dissipate excess performance. Following third stage burnout, the control system orients the vehicle to the proper attitude for fourth stage ignition. Following a long third stage coast phase (200-600 seconds depending upon the mission) the spacecraft plus fourth stage are spun up and separated from the expended third stage. The fourth stage is then ignited to provide the necessary velocity for injection into orbit. The spacecraft may or may not be separated from the expended fourth stage following burnout.

The present Scout vehicle third stage coast time is currently limited to 600 seconds, due to a thermal constraint on "C" section. This constraint restricts the spacecraft injection altitude to approximately 1100 kilometers; however, the coast time limitation may be relaxed after analysis of the specific mission requirements.

3.1 ORBITAL MISSION CAPABILITIES

Circular and elliptical orbit payload weight capabilities are presented in Figures II-2 through II-4 for due East launches from San Marco and Wallops Flight Center and polar launches for Vandenberg AFB. The San Marco launch site provides higher performance capability than either Wallops Flight Center or Vandenberg AFB since its location allows greater utilization of the earth's rotational velocity.

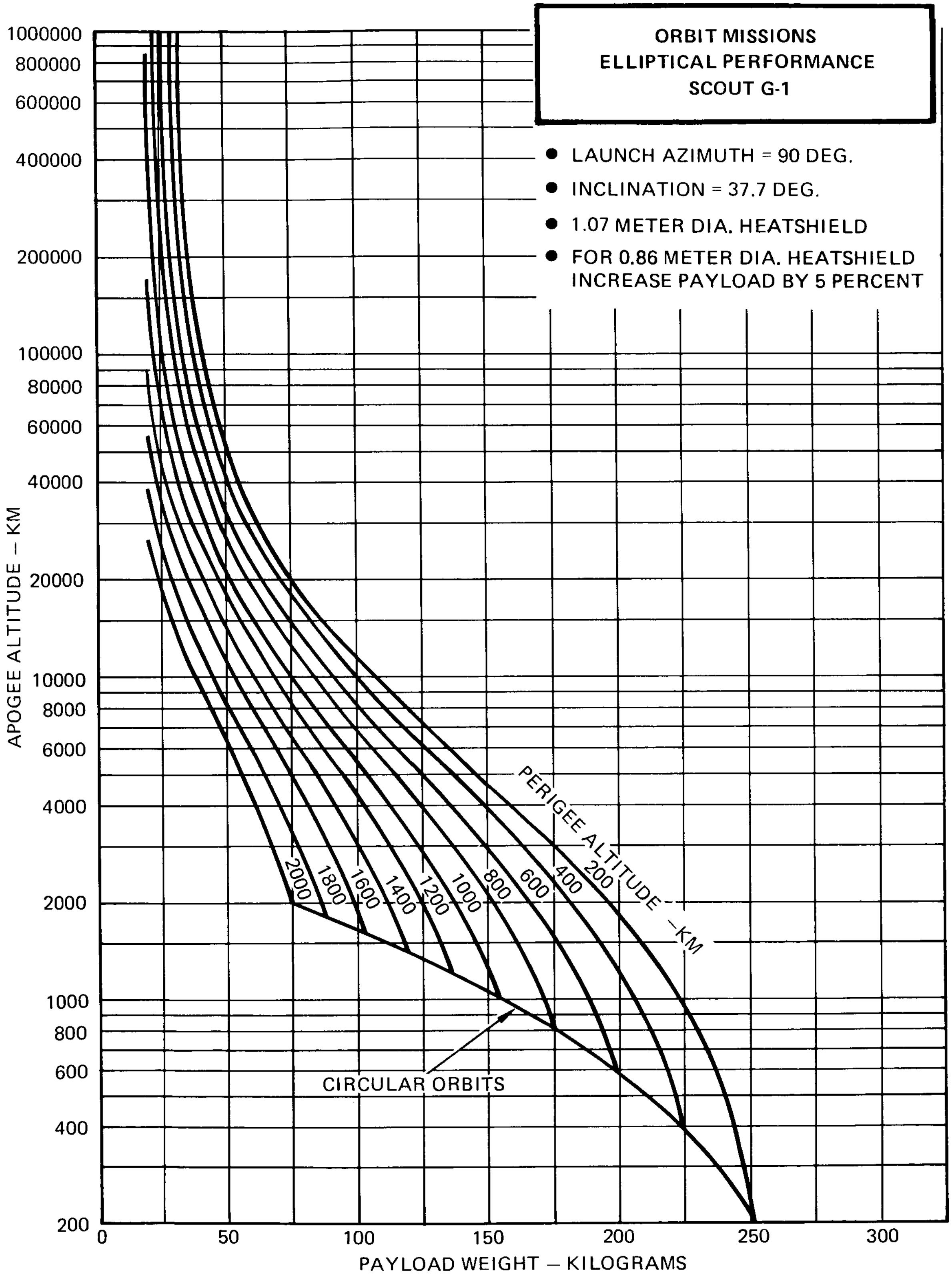

FIGURE II–2 ELLIPTICAL ORBIT PERFORMANCE – WFC

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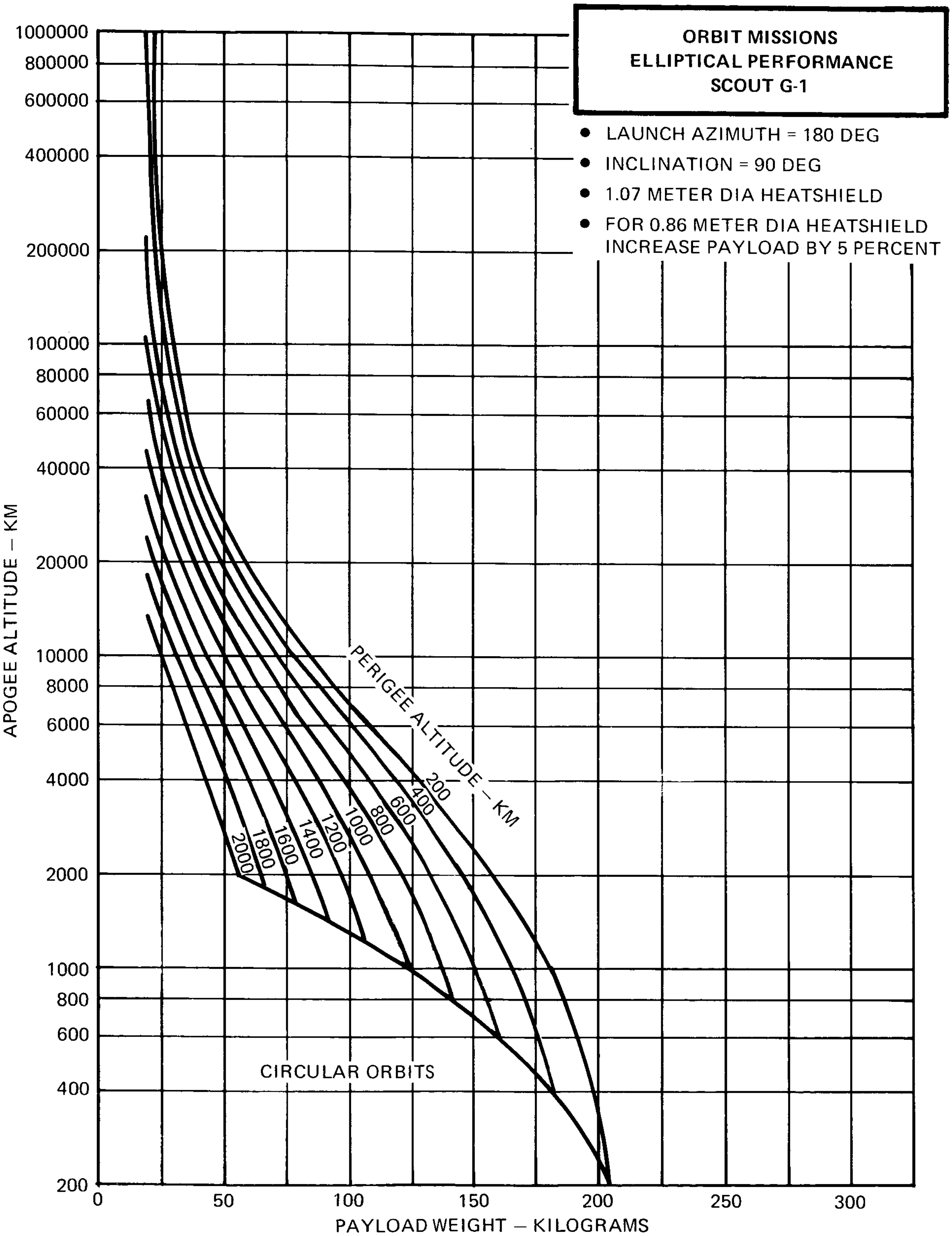

FIGURE II–3 ELLIPTICAL ORBIT PERFORMANCE – VANDENBERG AFB

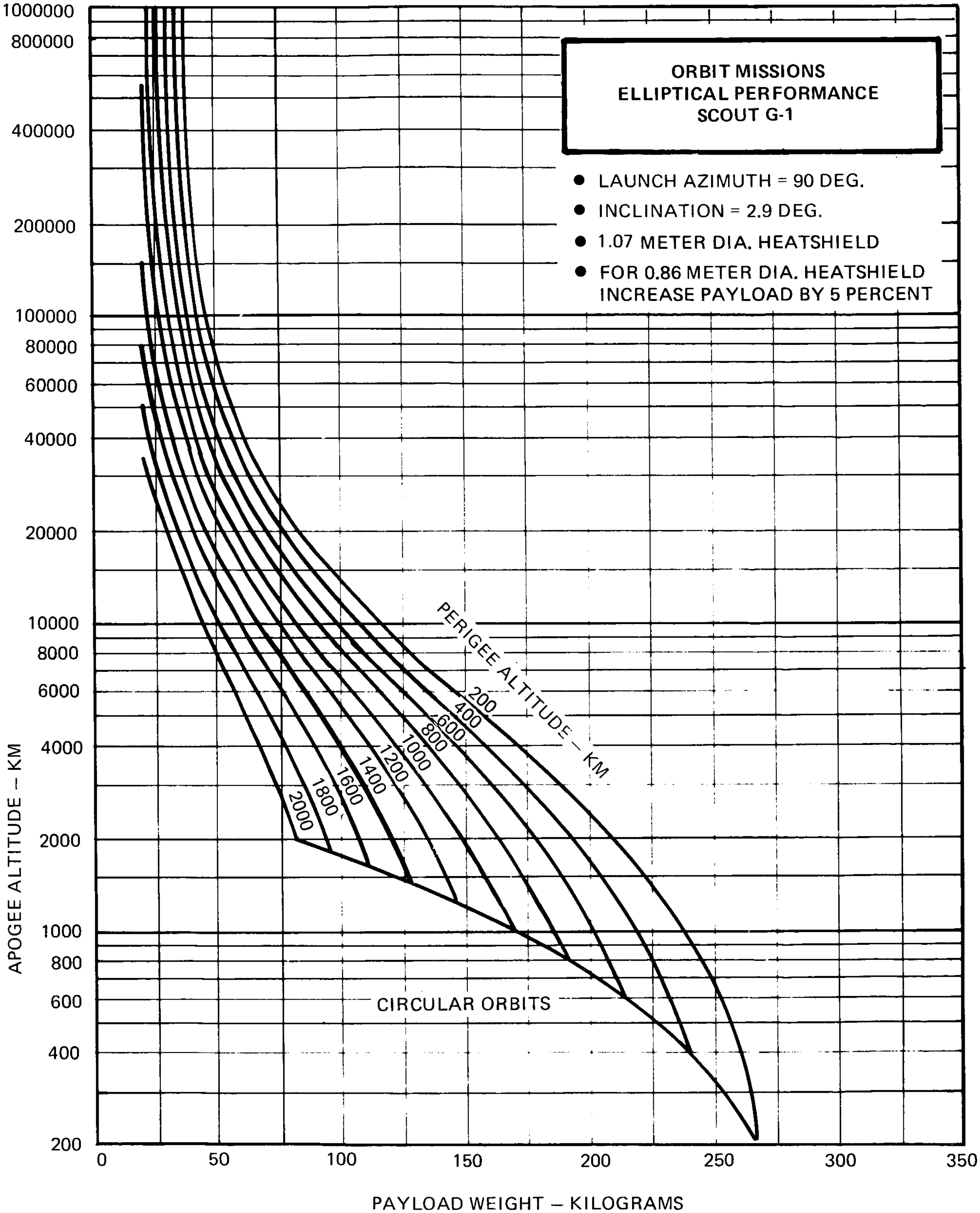

FIGURE II–4 ELLIPTICAL ORBIT PERFORMANCE – SAN MARCO – SCOUT G-1

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3.2 GROUND TRACK TO INJECTION

Geographic displays of the ground tracks to injection for three typical circular orbits are shown in Figures II-5, II-6, and II-7 for Wallops, VAFB, and San Marco launches, respectively. The ground tracks are presented as functions of initial launch azimuth. The launch azimuth limitations applicable to each launch site are also included.

3.3 EXPENDED STAGE IMPACT RANGE AND DISPERSION

Range safety considerations require a knowledge of the nominal impact points of the expended stages which fall back to earth. Impact points and predicted dispersion are calculated for each specific mission and are included as part of the Trajectory Data Package or Pre-Flight Brochure provided in the support of each Scout launch.

ORBIT MISSIONS
GROUND TRACK TO INJECTION
FOUR STAGE SCOUT

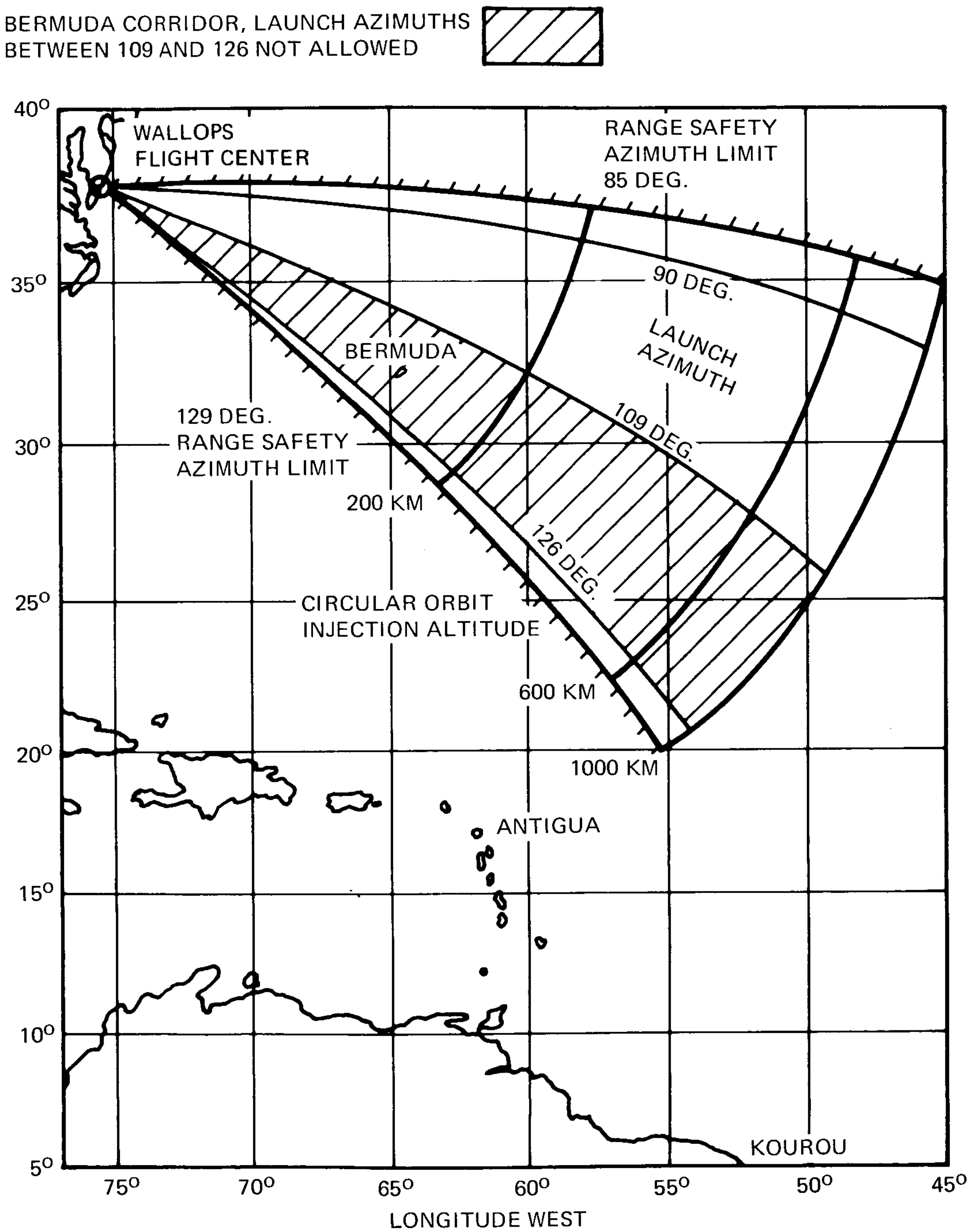

FIGURE II–5 GROUND TRACK TO INJECTION – WFC

ORBIT MISSIONS
GROUND TRACK TO INJECTION
FOUR STAGE SCOUT

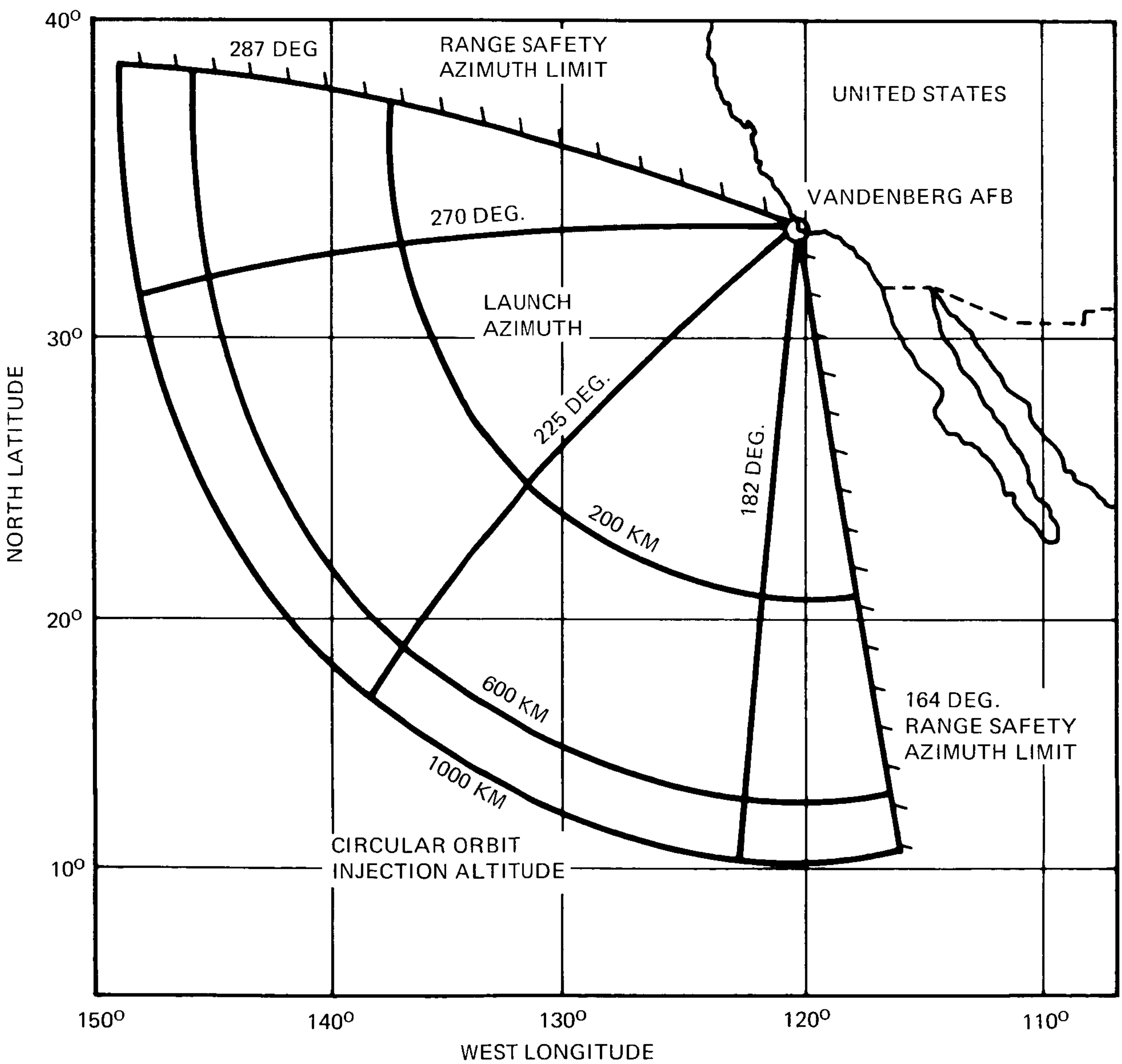

FIGURE II–6 GROUND TRACK TO INJECTION – VANDENBERG AFB

ORBIT MISSIONS
GROUND TRACK TO INJECTION
FOUR STAGE SCOUT

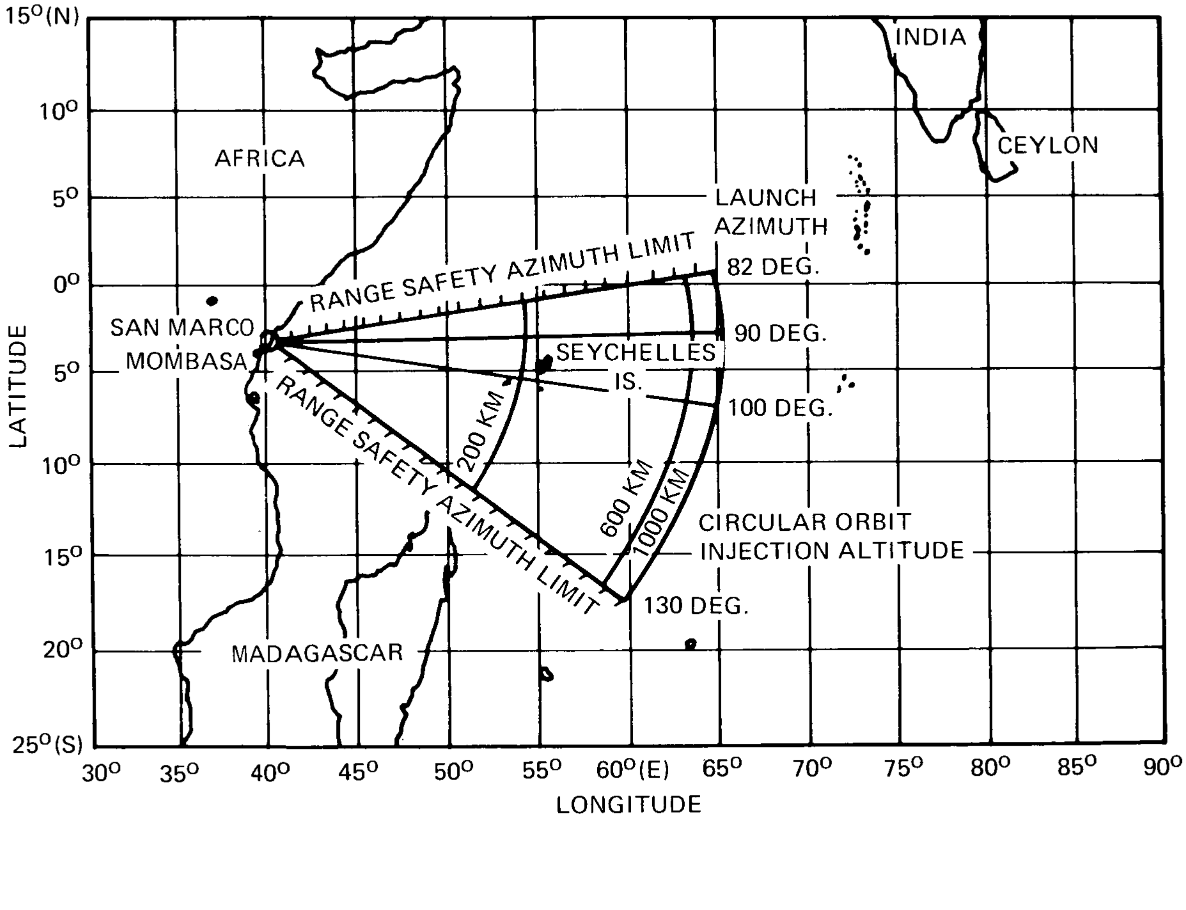

FIGURE II–7 GROUND TRACK TO INJECTION – SAN MARCO

3.4 ORBIT INCLINATION EFFECTS

The orbital mission performance data presented thus far are for a single orbit inclination for each launch site and are representative of the performance available. However, a large range of orbit inclinations can be achieved from the three launch sites. The range of orbit inclinations available and the resulting performance gains or losses are the subject of the following curves.

Figure II-8 shows the range of orbit inclinations which can be achieved from the three launch sites by direct injection, i.e., no plane change during boost operations. Inclination is presented as a function of the launch azimuth required to produce the inclination. Note that permissible launch azimuths from Wallops Flight Center exclude the corridor between azimuths of 109 and 126 degrees to avoid possible second stage impact inside the restricted area around the island of Bermuda. These restrictions are approximate, based on range safety considerations. Missions which require launch azimuths close to the specified limits will be scrutinized by the respective ranges to insure safe operations.

The Scout guidance system has the capability to command a yaw program prior to fourth stage ignition. The "yaw-torquing" technique may be utilized to perform a plane change. This method, sometimes referred to as a "dog-leg" maneuver, may be used to either increase or decrease the orbit inclination obtained by the direct injection method for a given launch azimuth.

Figure II-9 presents representative payload weight losses incurred for other than due East launches (37.7 degree inclination) from Wallops Flight Center for the four stage Scout configuration. The payload weight-inclination relationship for direct injection and "dog-leg" maneuvers to the right and left are shown for typical circular orbits. Inclinations greater than 51.7 degrees are achieved by using a launch azimuth of 129 degrees combined with a "dog-leg" right maneuver. Inclinations less than 37.7 degrees are achieved by using a launch azimuth greater than 90 degrees combined with a "dog-leg" left maneuver. Since more than one combination of launch azimuth and "dog-leg" left maneuver produce a given inclination less than 37.7 degrees, the combination which provides the minimum payload weight loss would be selected.

Those inclinations between 41.5 degrees and 49.5 degrees which are within the Bermuda Corridor can only be obtained by using "dog-leg" maneuvers. For those missions, a launch azimuth of 109 degrees combined with a "dog-leg" right maneuver, or a launch azimuth of 126 degrees combined with a "dog-leg" left maneuver would be used. The choice would depend on which method resulted in the minimum payload weight loss.

The performance gains or losses as a function of inclination resulting from non-polar launches from VAFB are shown in Figure II-10 for typical circular orbits. The data presented are applicable to the four stage Scout configuration. This figure shows the payload-inclination relationships for prograde, retrograde and "dog-leg" orbits. Both prograde and retrograde orbits can be achieved from the VAFB by direct injection. Prograde orbits can be achieved by launching on southeasterly headings and retrograde orbits can be achieved by launching on westerly headings. Prograde orbits result in gains in payload capability where as retrograde orbits result in losses in payload

ORBIT MISSION AZIMUTH-INCLINATION RELATIONSHIP

- DIRECT INJECTION
- CIRCULAR ORBITS
- LAUNCH AZIMUTH LIMITS
  - SAN MARCO = 82 THROUGH 130 DEGREES
  - WALLOPS FLIGHT CENTER = 85 THROUGH 109 DEGREES
    126 THROUGH 129 DEGREES
  - VANDENBERG AFB = 164 THROUGH 287 DEGREES

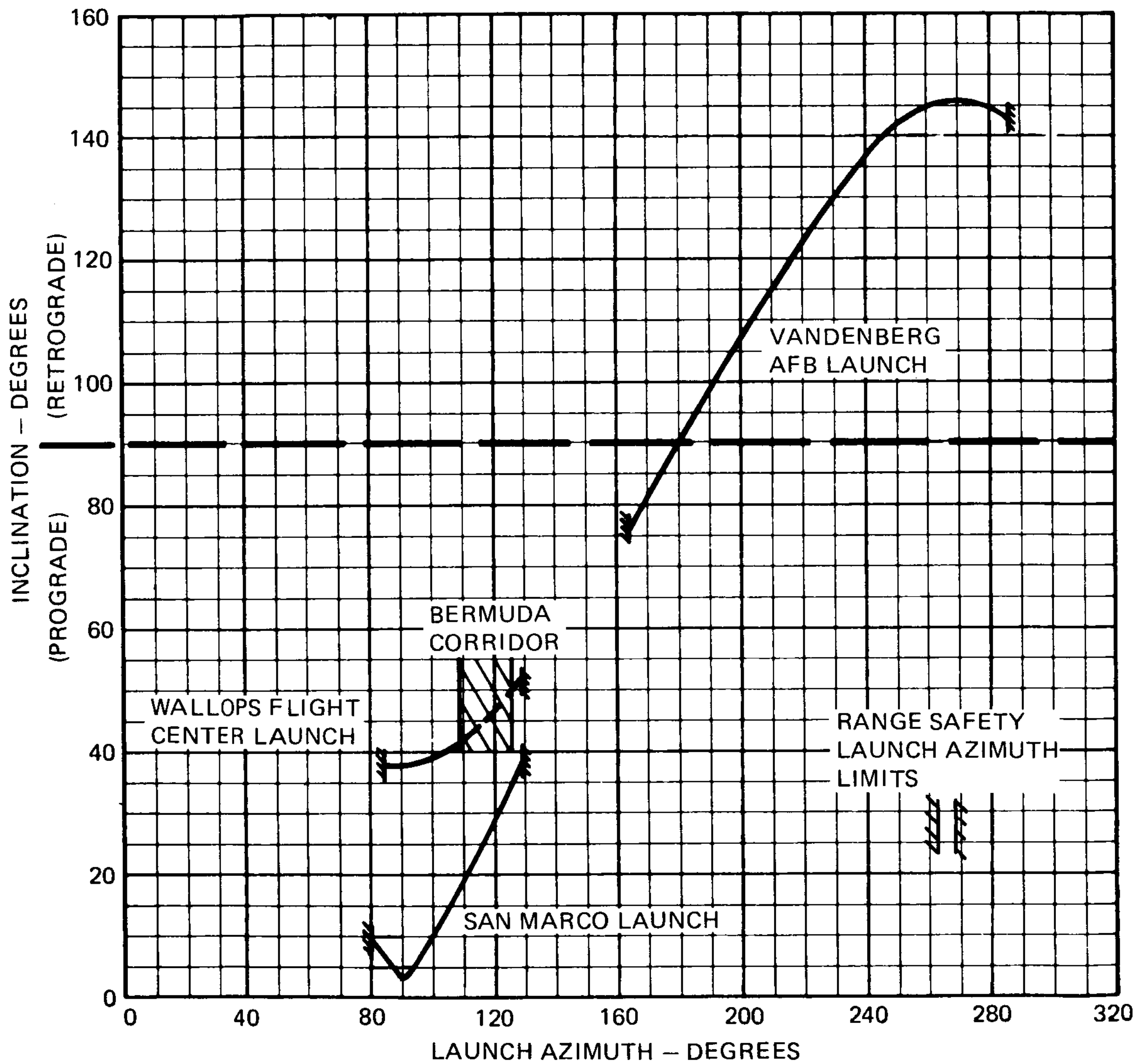

FIGURE II–8 AZIMUTH-INCLINATION RELATIONSHIP (CIRCULAR ORBITS)

ORBIT MISSIONS
PAYLOAD-INCLINATION TRADE
SCOUT G-1

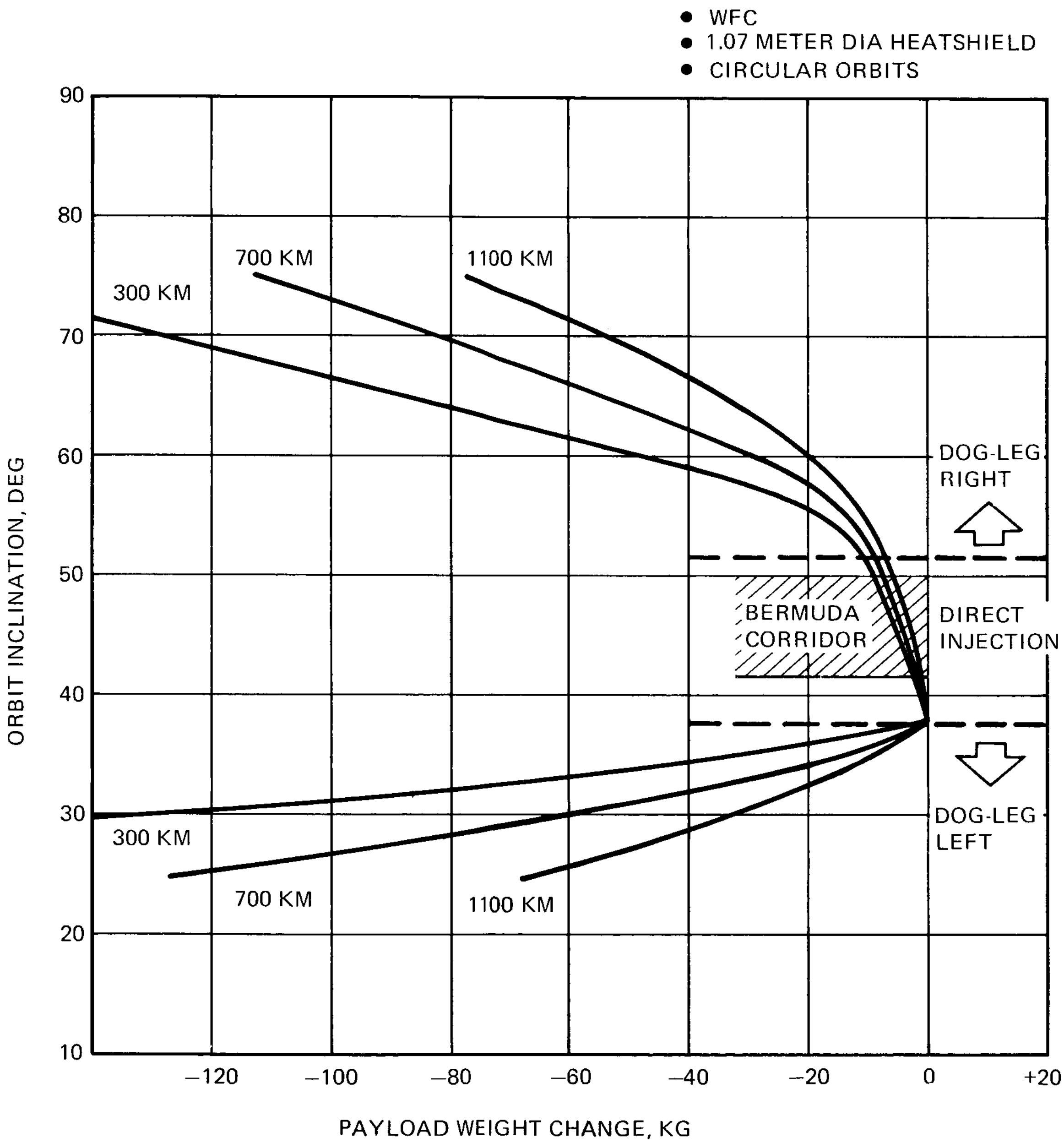

FIGURE II–9 PAYLOAD INCLINATION RELATIONSHIP – WFC

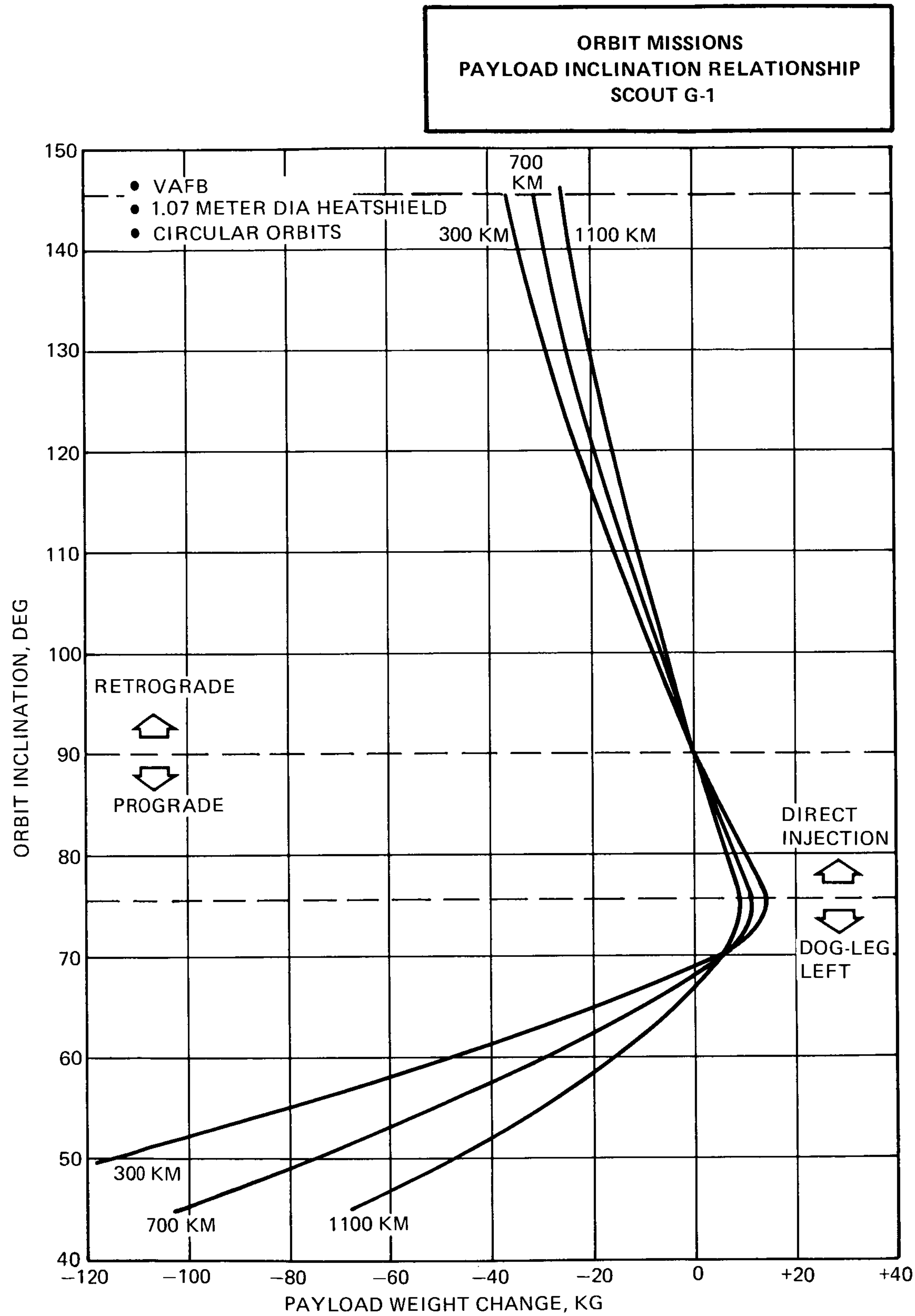

FIGURE II –10 PAYLOAD INCLINATION RELATIONSHIP – VANDENBERG AFB

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ORBIT MISSIONS
PAYLOAD – INCLINATION RELATIONSHIP
FOUR STAGE SCOUT

- CIRCULAR ORBITS AS NOTED
- 1.07 METER DIA HEATSHIELD

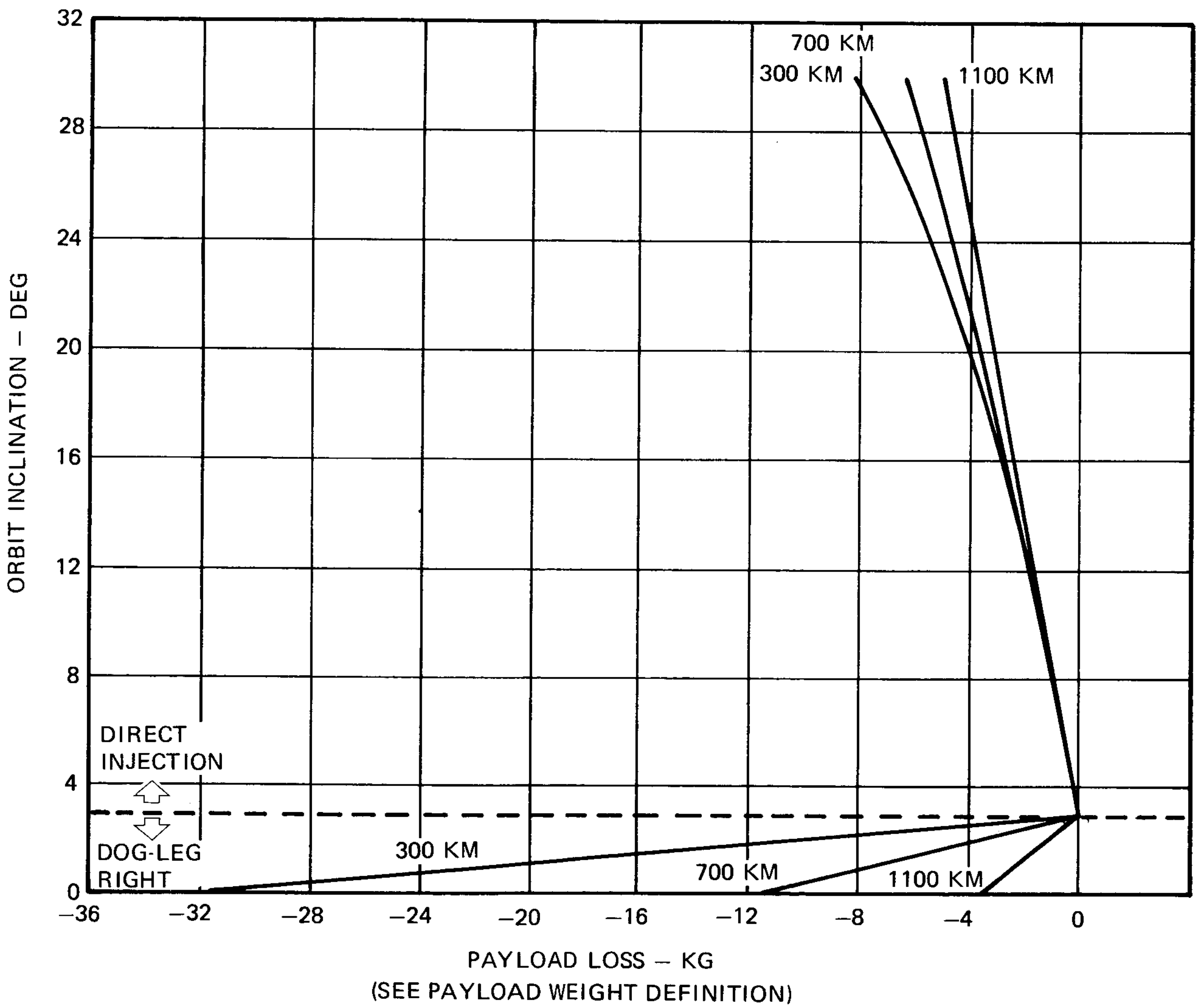

FIGURE II–11 PAYLOAD-INCLINATION RELATIONSHIP – SAN MARCO

capability when compared with the basic performance capability for polar orbits. Within the launch azimuth limitations dictated by range safety considerations, prograde orbits with inclinations between approximately 90 and 76.5 degrees can be achieved. The inclination range for retrograde orbits is between approximately 90 and 145 degrees.

By utilizing the "yaw-torquing" technique in "dog-leg" maneuvers to the left from a launch azimuth of 164 degrees, inclinations less than 76.5 degrees can be obtained. It can be seen from the figure that some prograde orbits achieved by a "dog-leg" maneuver result in a payload weight gain compared to a retrograde orbit achieved by direct injection with the same inclination magnitude. For instance, a 65 degree prograde orbit, which requires a "dog-leg" maneuver, results in more payload weight capability than a 115 degree retrograde orbit which is achieved by direct injection.

The performance losses resulting from launches at a launch azimuth other than due east from San Marco are given in Figure II-11 as a function of inclination angle. Inclinations less than 2.9 degrees can be obtained only by using "dog-leg" maneuvers.

3.5 HEATSHIELD EJECTION ALTITUDE EFFECTS

Orbital mission performance in this chapter is based on ejection of the heatshield at nominal altitudes. Nominally, the heatshield is ejected approximately 1.7 seconds prior to ignition of the third stage. Higher ejection altitudes are obtainable for missions having more critical payload environmental constraints. The payload weight penalty for higher than nominal heatshield ejection altitudes isshown in Figure II-12 for several typical circular polar orbits from VAFB.

3.6 ACCURACY

Figure II-13 presents typical one standard deviation in inclination as a function of nominal inclination and launch azimuth for the Wallops Flight Center VAFB, and San Marco launch sites. These deviations are based on Scout flight history and represent an average spin rate of 157 revolutions per minute. The deviations will increase for spin rates below the average and decrease for spin rates above the average. Care should be taken to avoid misinterpretation on this figure. On a portion of the curve for San Marco and WFC, two values of standard deviations are shown for a single inclination since two different launch azimuths produce the same inclination. Therefore, to use this portion of the curve, the launch azimuth as well as the inclination must be known.

- CIRCULAR ORBITS
- VAFB POLAR LAUNCHES
- 1.07 METER DIA HEATSHIELD
- NOMINAL HEATSHIELD EJECTION OCCURS APPROXIMATELY 2.0 SEC BEFORE THIRD STAGE IGNITION TIME

HEATSHIELD EJECTION ALTITUDE PAYLOAD RELATIONSHIP FOUR STAGE SCOUT

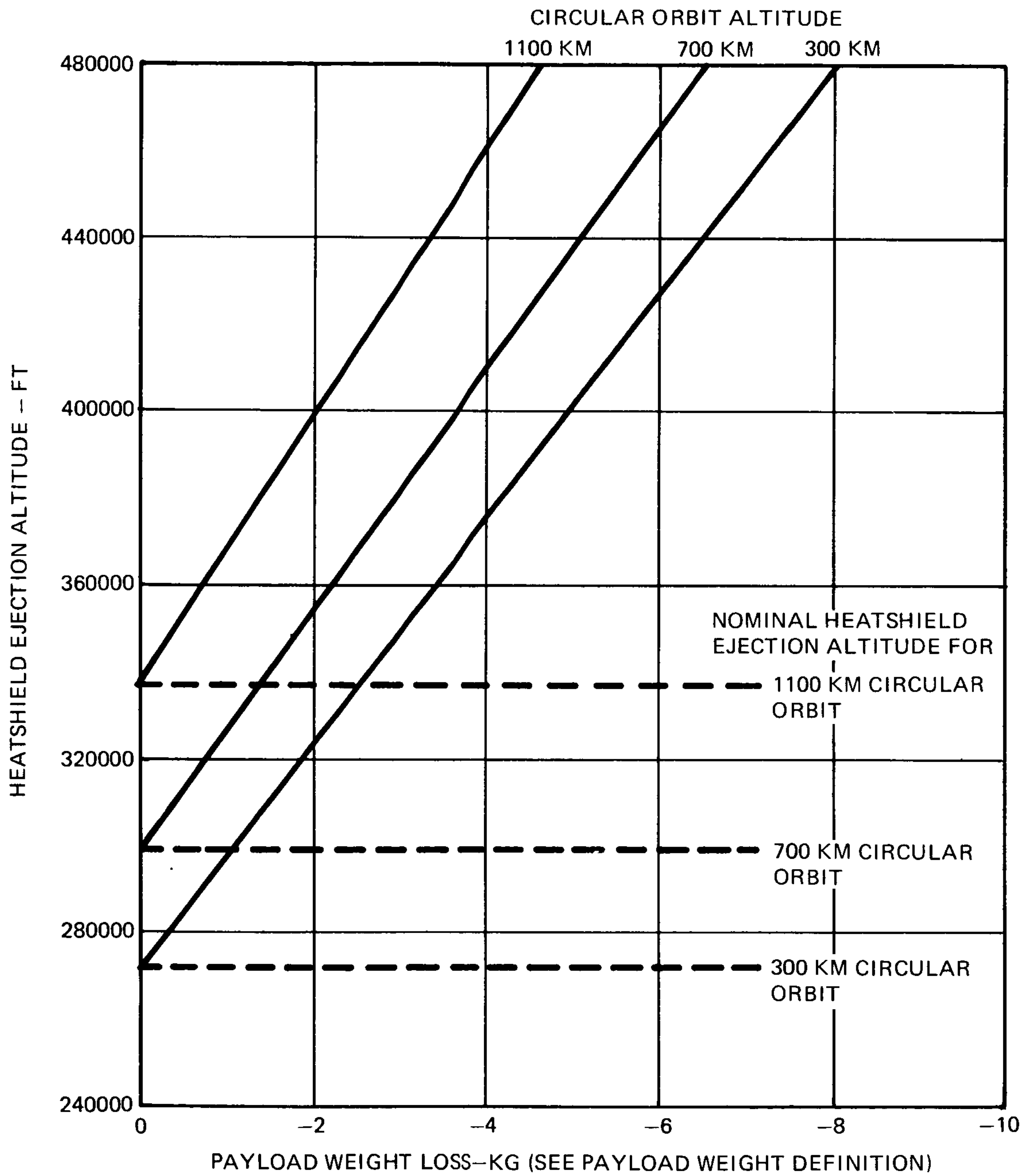

FIGURE II–12 **HEATSHIELD EJECTION ALTITUDE – PAYLOAD RELATIONSHIP VANDENBERG AFB**

ORBIT MISSIONS
INCLINATION ACCURACY
FOUR STAGE SCOUT

- 550 KILOMETER CIRCULAR ORBIT
- RANGE SAFETY LIMITS
- BERMUDA CORRIDOR = 109 DEG. to 126 DEG. LAUNCH AZIMUTH
- BASED ON SCOUT FLIGHT EXPERIENCE

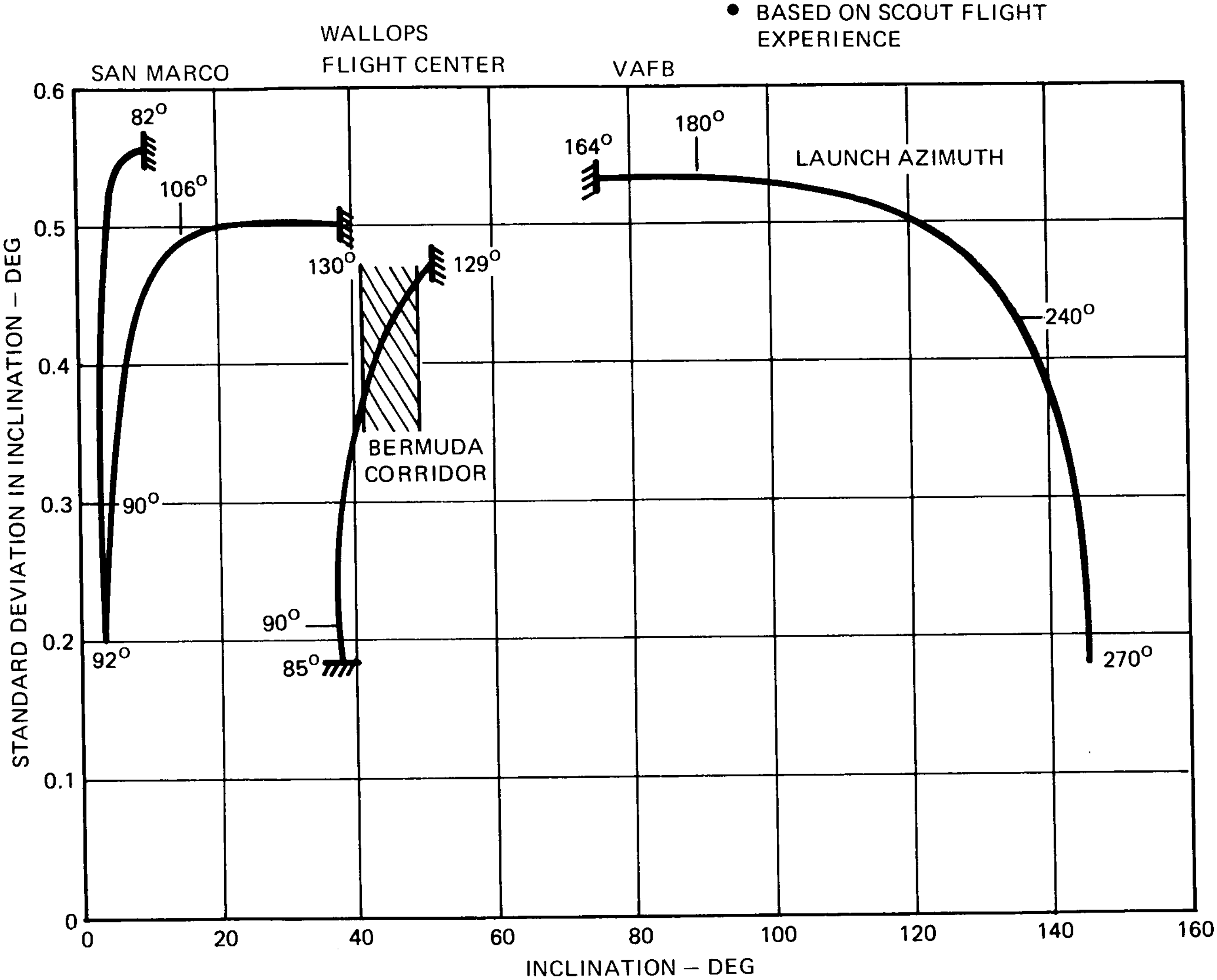

FIGURE II–13 INCLINATION ACCURACY – ONE STANDARD DEVIATION

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Scout accuracy of apogee and perigee altitude are presented in Figure II-14 (sheets 1 through 4). These figures show isoprobability contours of apogee-perigee deviations for probability values of 0.95 and 0.75 and for injection altitudes of 300, 550, 800 and 1100 kilometers. Contours are shown for circular orbits and for three elliptical orbits with apogee ranging up to 2700 km greater than perigee. Isoprobability contours show the distribution of possible combinations of apogee-perigee deviations consistent with a specified probability value (i.e. for a 0.95 probability contour, the apogee-perigee deviations of an obtained orbit will fall within the contour 95 percent of the time). When using these contours, the combination of apogee-perigee deviations must be considered rather than just the deviation in apogee or perigee alone. An example use of the contours follows: for a probability value of 0.95 and an orbit with a perigee altitude of 300 km and an apogee altitude of 400 km, the maximum apogee deviations are +198 km and −110 km, with associated perigee deviations of +10 km and −105 km, respectively. An improper use of this contour is to say that the maximum deviations in apogee altitude are +198 km and −110 km regardless of the perigee deviations (e.g. a +198 km apogee deviation does not exist at the maximum perigee deviation of −105 km)

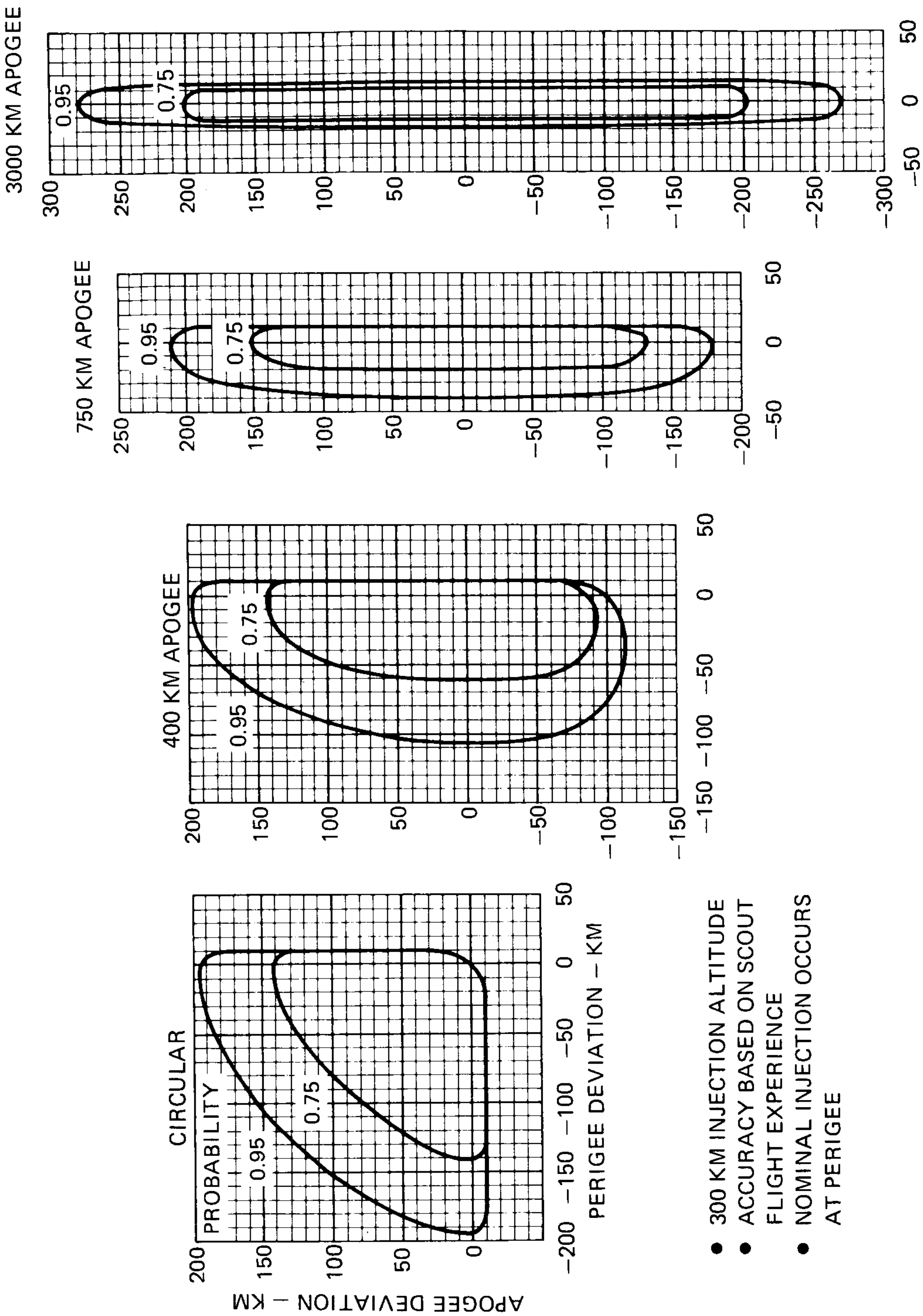
For orbits other than those shown, data can be obtained by cross-plotting the information from the contours.

As shown by these figures, the trend of perigee deviations with increasing nominal apogee altitude indicates a significant reduction in perigee deviation for slightly elliptical orbits when compared with circular orbits.

Figure II-15 presents one standard deviation of argument of perigee as a function of apogee altitude for a 300 km injection altitude orbit. Argument of perigee accuracy degrades only slightly for higher injection altitudes.

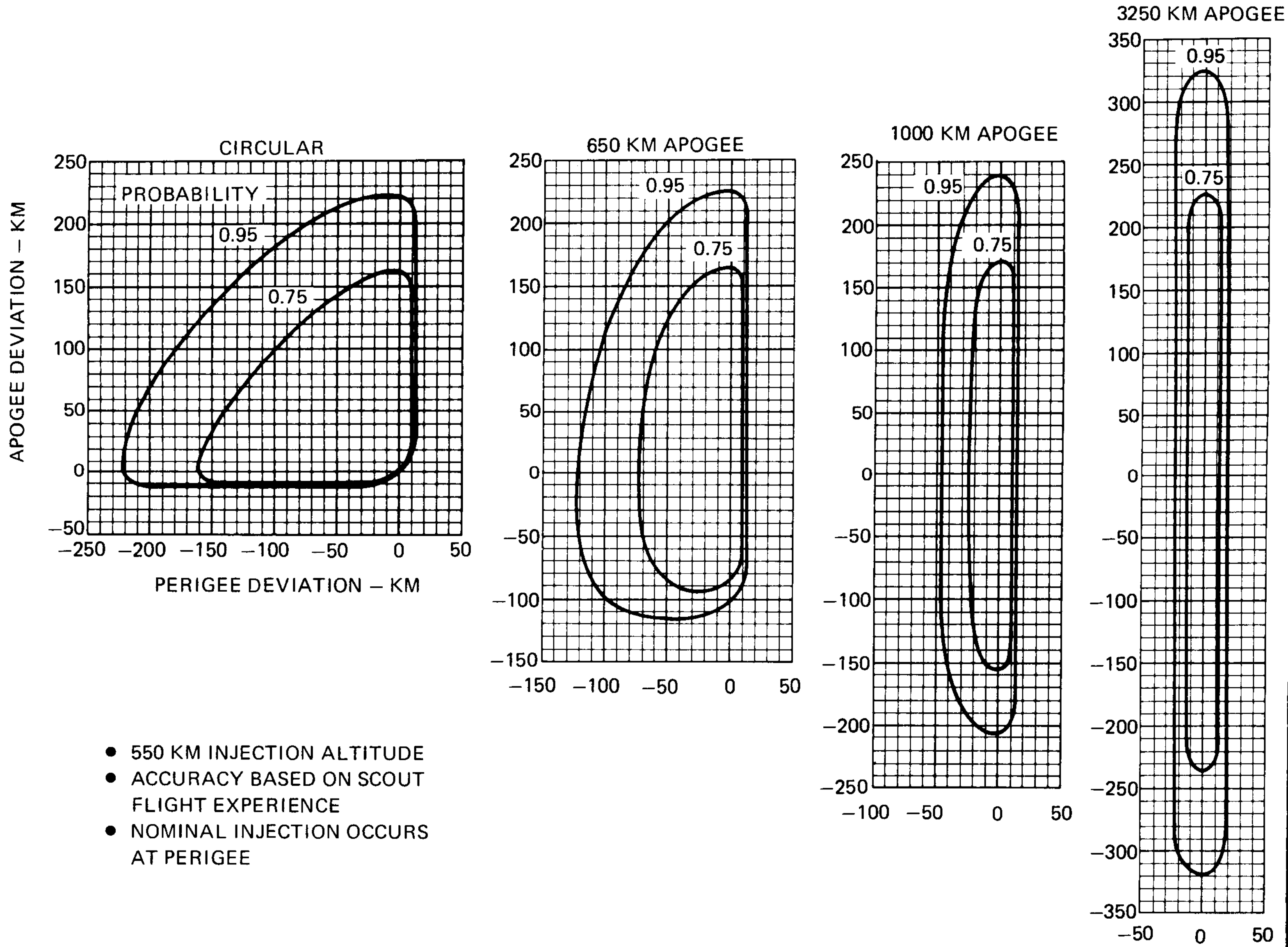
Figure II-16 presents nodal drift rate accuracy for sun-synchronous circular orbits. As shown for a 300 km circular orbit, the probability is 0.95 that the achieved deviation in nodal drift rate will be between −0.155 and +0.155 degree per day.

ORBIT MISSIONS
ORBITAL ACCURACY



- 300 KM INJECTION ALTITUDE
- ACCURACY BASED ON SCOUT FLIGHT EXPERIENCE
- NOMINAL INJECTION OCCURS AT PERIGEE

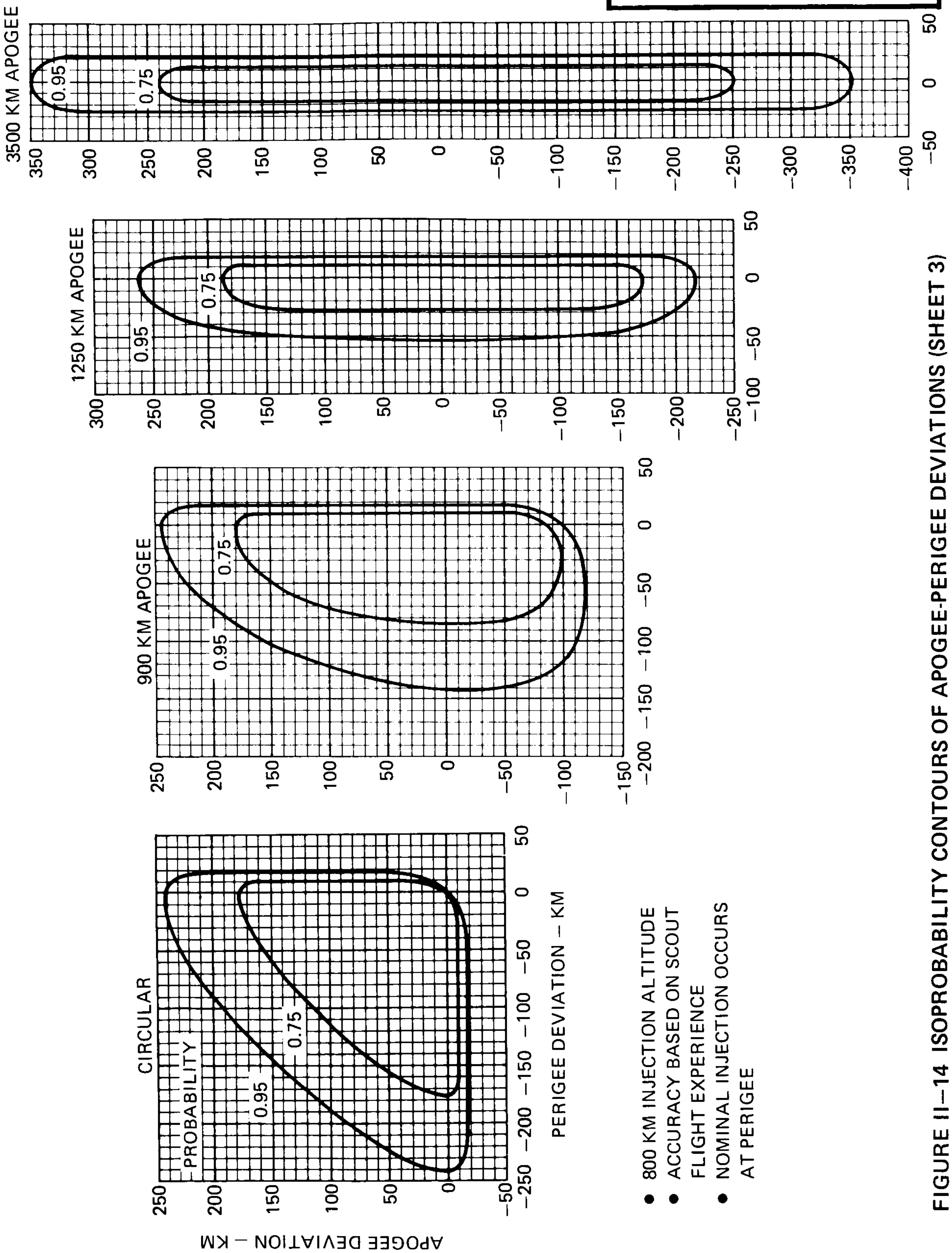
FIGURE II–14 ISOPROBABILITY CONTOURS OF APOGEE-PERIGEE DEVIATIONS (SHEET 1)



- 550 KM INJECTION ALTITUDE
- ACCURACY BASED ON SCOUT FLIGHT EXPERIENCE
- NOMINAL INJECTION OCCURS AT PERIGEE

FIGURE II–14 ISOPROBABILITY CONTOURS OF APOGEE-PERIGEE DEVIATIONS (SHEET 2)

ORBIT MISSIONS
ORBITAL ACCURACY



- 800 KM INJECTION ALTITUDE
- ACCURACY BASED ON SCOUT FLIGHT EXPERIENCE
- NOMINAL INJECTION OCCURS AT PERIGEE

FIGURE II–14 ISOPROBABILITY CONTOURS OF APOGEE-PERIGEE DEVIATIONS (SHEET 3)

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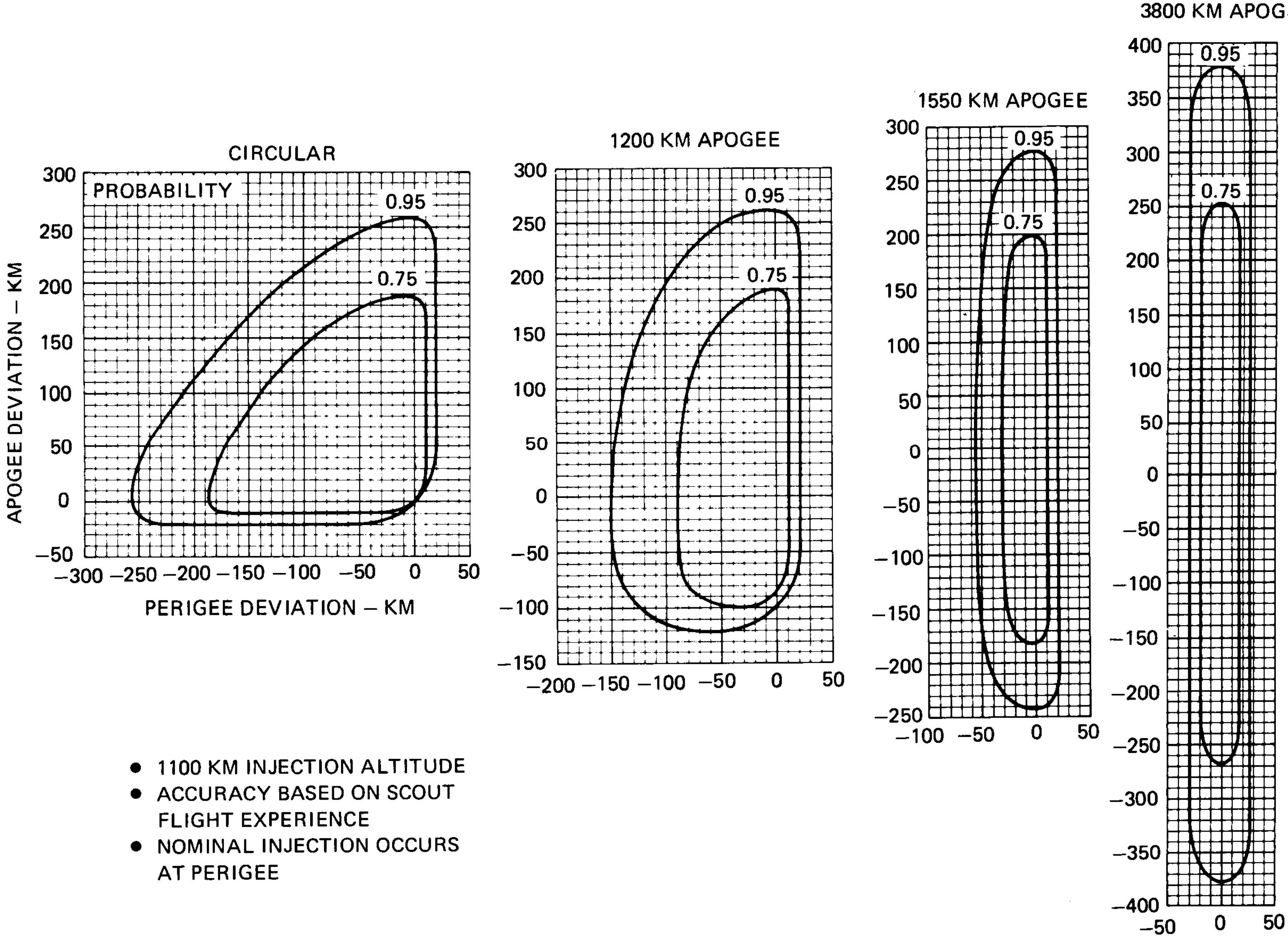

- 1100 KM INJECTION ALTITUDE
- ACCURACY BASED ON SCOUT FLIGHT EXPERIENCE
- NOMINAL INJECTION OCCURS AT PERIGEE

FIGURE II–14 ISOPROBABILITY CONTOURS OF APOGEE-PERIGEE DEVIATIONS (SHEET 4)

ARGUMENT OF PERIGEE ACCURACY
300 KM INJECTION ALTITUDE

- INJECTION OCCURS AT PERIGEE
- BASED ON SCOUT FLIGHT EXPERIENCE
- ACCURACY DEGRADES SLIGHTLY WITH INCREASING INJECTION ALTITUDE

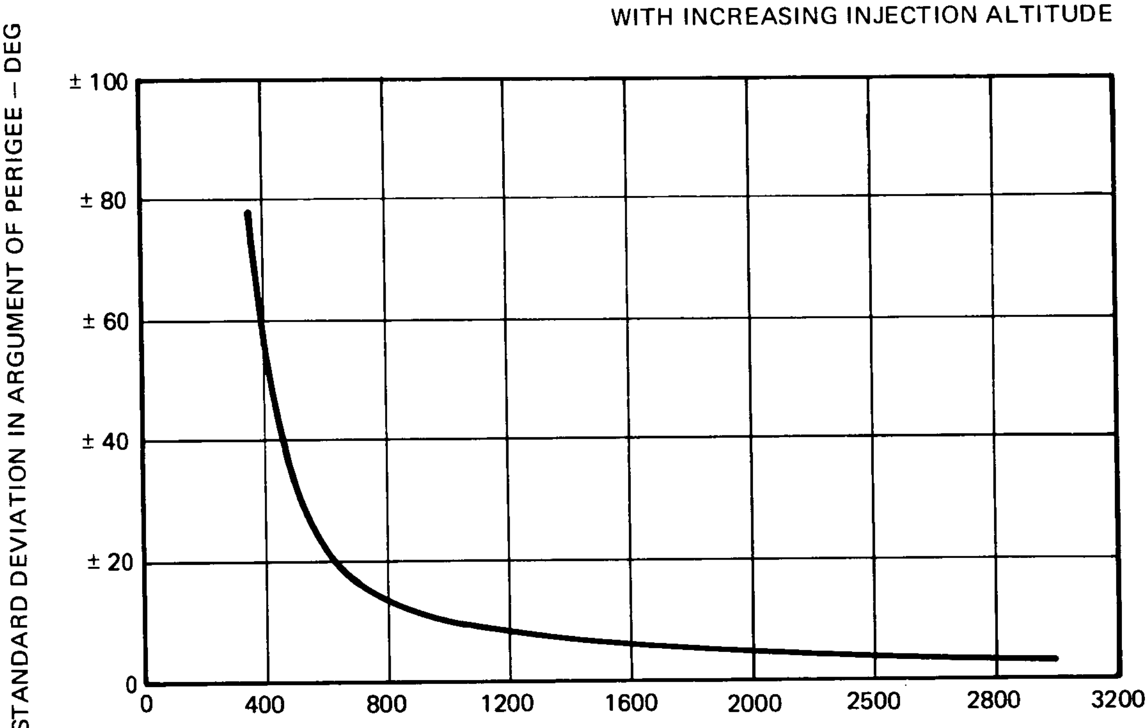

FIGURE II–15 ARGUMENT OF PERIGEE ACCURACY

NODAL DRIFT RATE ACCURACY
SUN-SYNCHRONOUS ORBITS

• BASED ON SCOUT FLIGHT EXPERIENCE

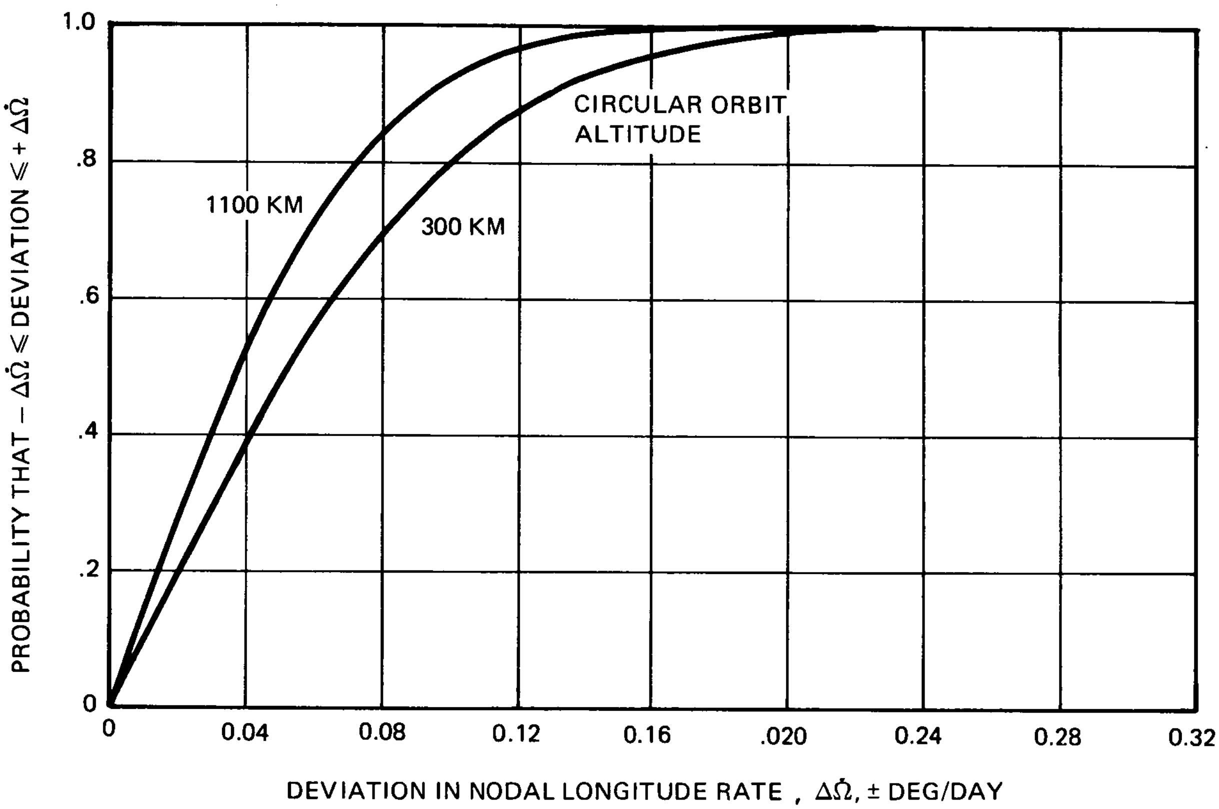

FIGURE II–16 NODAL DRIFT RATE ACCURACY

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3.6.1 Summary of Scout Orbital Accuracy

Orbital accuracy, both predicted and measured, is summarized in Figure II-17. For each launch, this figure shows the predicted 0.997 probability for apogee, perigee and inclination angle deviations and the post-flight values measured for each parameter.

Scout orbital launches are summarized in Table 11-3 which correlates launch number with vehicle number and launch site, gives predicted (pre-flight) and measured (post-flight) orbital parameters, and specifies vehicle configuration for each launch.

Although orbit lifetime is not related specifically to Scout performance, it is often a factor in orbit mission planning. For this reason, parametric orbit lifetime data is presented for use in preliminary mission planning. These data are applicable to desired nominal orbits.

3.7 ORBIT LIFETIME

Orbit lifetime is dependent upon orbit altitude, inclination, argument of perigee, launch date, launch time of day and ballistic coefficient of the spacecraft. Of these, only orbit altitude, launch date and ballistic coefficient are significant in calculating lifetime data for preliminary mission planning. Launch date is significant because of the time varying earth's atmospheric density, which is largely determined by solar activity. In order to simplify lifetime data presentation, average predicted solar activity was used for time spans of two years. This simplification results in a set of lifetime curves, Figure 11-18, for launch dates of 1 January 1979 through 1 January 1981. This figure is applicable for launch dates covering three years since the average solar activity is the same during each year. These data are shown for a ballistic coefficient of 1 kg/m^2. Since constant solar activity was assumed, the orbit lifetime for a specific spacecraft is obtained by multiplying the given lifetime by the ballistic coefficient of the spacecraft. All lifetime data are approximate and should not be used for lifetimes greater than two years from the launch date.

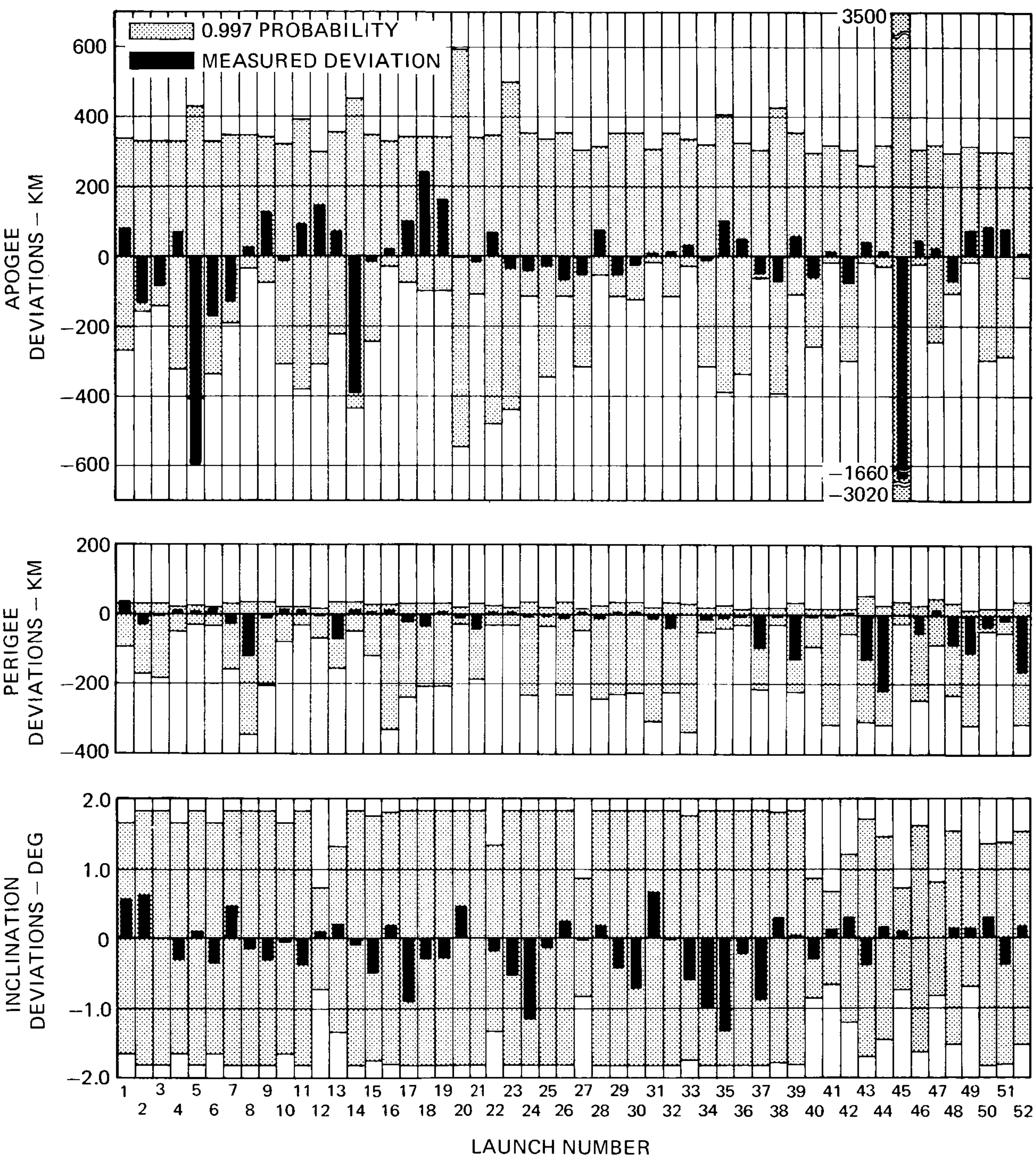

FIGURE II–17 SCOUT ORBITAL ACCURACY SUMMARY (SHEET 1)

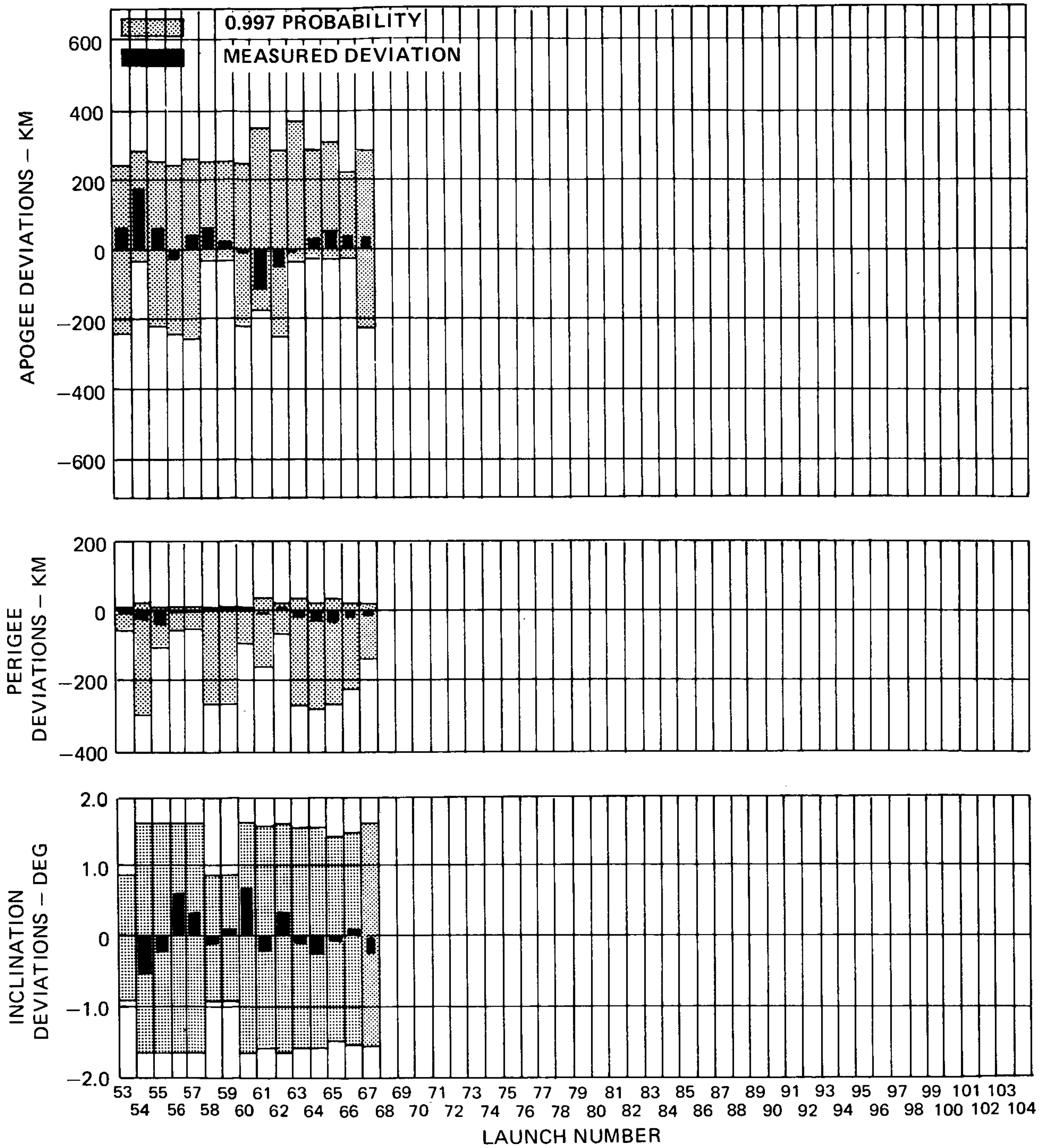

FIGURE II–17 SCOUT ORBITAL ACCURACY SUMMARY (SHEET 2)

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TABLE II–3 SCOUT LAUNCH SUMMARY (SHEET 1)

DECEMBER 1962 – JANUARY 1980

LAUNCH NUMBER	LAUNCH SITE	SCOUT VEHICLE NUMBER	APOGEE KILOMETERS		PERIGEE KILOMETERS		INCLINATION ANGLE DEGREES	
			PREFLIGHT	POST-FLIGHT	PREFLIGHT	POST-FLIGHT	PREFLIGHT	POST-FLIGHT
1	WFC	S–115 (1)	1099.2	1183.4	728.8	763.2	51.43	52.00
2	VAFB	S–118 (1)	872.1	734.5	734.5	704.9	90.00	90.62
3	VAFB	S–120 (1)	851.2	766.7	736.2	731.9	90.01	90.01
4	WFC	S–113 (2)	1246.2	1315.8	409.3	421.5	50.02	49.71
5	VAFB	S–122 (2)	2993.6	2398.2	596.5	604.1	78.50	78.61
6	WFC	S–127 (1)	1524.2	1355.7	281.5	296.3	52.03	51.67
7	VAFB	S–125 (2)	1095.5	964.0	888.6	861.9	90.01	90.48
8	VAFB	S–134 (2)	1004.2	1028.6	996.2	877.8	80.02	79.87
9	VAFB	S–123 (2)	962.3	1089.9	910.3	895.3	80.00	79.69
10	WFC	S–133 (2)	999.7	986.9	458.4	471.9	51.99	51.94
11	VAFB	S–135 (2)	2410.6	2503.9	527.8	536.7	81.72	81.35
12	WFC	S–137 (2)	677.1	822.3	213.5	208.9	37.69	37.78
13	WFC	S–136 (2)	1253.2	1324.4	1015.5	943.4	40.98	41.17
14	WFC	S–131 (4)	2826.3	2434.1	1134.4	1144.7	69.36	69.25
15	WFC	S–138 (2)	1007.5	894.7	710.6	716.0	60.19	59.71
16	VAFB	S–139 (2)	748.8	769.0	744.5	756.2	75.70	75.88
17	VAFB	S–140 (3)	991.6	1092.7	937.7	918.0	90.00	89.10
18	VAFB	S–142 (3)	984.7	1223.6	902.1	870.6	90.00	89.71
19	VAFB	S–143 (3)	973.8	1136.0	892.8	900.1	90.00	89.73
20	VAFB	S–145 (4)	5751.4	5745.5	370.6	361.5	82.00	82.46
21	VAFB	S–146 (3)	1004.9	990.1	910.8	868.8	90.00	90.00
22	WFC	S–147 (4)	4670.6	4738.0	649.3	651.0	40.99	40.82
23	VAFB	S–148 (4)	4523.0	4488.9	366.1	371.1	82.00	81.47
24	VAFB	S–149 (3)	1154.9	1115.5	1067.3	1058.4	90.00	88.85
25	VAFB	S–150 (4)	1637.2	1607.9	330.6	326.7	82.10	81.98
26	VAFB	S–154 (3)	1157.7	1091.6	1068.8	1057.7	90.00	90.25
27	SM	S–153 (4)	800.6	749.1	216.7	218.7	2.92	2.89
28	VAFB	S–155 (3)	536.2	611.2	517.4	507.3	80.00	80.17
29	VAFB	S–156 (3)	1166.6	1112.1	1079.2	1081.6	90.00	89.58
30	VAFB	S–157 (3)	1148.8	1124.5	1047.9	1049.7	90.00	89.29
31	VAFB	S–158 (4)	440.0	448.0	430.4	417.1	90.00	90.66
32	VAFB	S–162 (3)	1162.1	1150.5	1076.4	1038.6	90.00	89.98
33*	WFC	S–160 (4)	861.0	892.1	857.8	518.4	60.00	59.41
34	VAFB	S–161 (4)	1099.7	1087.3	352.6	338.7	98.20	97.20
35	VAFB	S–165 (4)	2441.3	2541.9	697.8	687.8	81.99	80.67

CONFIGURATIONS:

(1) ALGOL II, CASTOR I, ANTARES II, ALTAIR I

(2) ALGOL II, CASTOR I, ANTARES II, ALTAIR II

(3) ALGOL II, CASTOR II, ANTARES II, ALTAIR II

(4) ALGOL II, CASTOR II, ANTARES II, ALTAIR III

(5) ALGOL IIIA, CASTOR IIA, ANTARES IIA, ALTAIR IIIA

(6) ALGOL IIIA, CASTOR IIA, ANTARES IIB, ALTAIR IIIA, ALCYONE IA

(7) ALGOL IIIA, CASTOR IIA, ANTARES IIB, ALTAIR IIIA

(8) ALGOL IIIA, CASTOR IIA, ANTARES IIIA, ALTAIR IIIA

* FIRST STAGE ANOMALY

** PITCH PROGRAM ANOMALY

ORBITAL MISSION PERFORMANCE

TABLE II–3 SCOUT LAUNCH SUMMARY (SHEET 2)

DECEMBER 1962 – JANUARY 1980

LAUNCH NUMBER	LAUNCH SITE	SCOUT VEHICLE NUMBER	APOGEE KILOMETERS		PERIGEE KILOMETERS		INCLINATION ANGLE DEGREES	
			PREFLIGHT	POST-FLIGHT	PREFLIGHT	POST-FLIGHT	PREFLIGHT	POST-FLIGHT
36	VAFB	S–167(4)	1499.7	1547.7	272.6	268.2	94.00	93.77
37	VAFB	S–172(4)	440.4	393.0	400.0	305.0	86.00	85.13
38	VAFB	S–169(4)	3228.2	3156.4	397.8	395.2	102.67	102.96
39	VAFB	S–176(3)	1175.8	1230.8	1089.5	962.8	90.00	90.02
40	WFC	S–174(4)	593.6	535.0	314.7	307.2	37.69	37.40
41	SM	S–175(4)	560.6	571.5	539.7	532.3	2.91	3.04
42	SM	S–173(4)	800.1	723.0	213.9	222.2	2.90	3.20
43	WFC	S–177(4)	598.8	639.5	566.5	442.4	51.43	51.05
44	WFC	S–180(4)	900.8	913.6	899.9	684.9	50.00	50.16
45	SM	S–163(4)	2858.2	2691.2	222.4	220.9	3.45	3.58
46	VAFB	S–183(4)	550.0	599.3	549.9	484.7	83.00	83.00
47	WFC	S–184(5)	791.2	822.6	478.7	503.9	37.69	37.69
48	VAFB	S–182(4)	915.1	847.4	832.6	750.7	90.00	90.13
49	SM	S–170(5)	555.3	631.8	554.1	444.3	1.77	1.90
50	VAFB	S–185(5)	1099.7	1184.2	280.0	251.5	90.80	91.11
51	VAFB	S–181(5)	797.9	878 3	239.9	229.8	97.20	96.95
52	VAFB	S–178(3)	1151.8	1153.6	1074.1	907.3	90.02	90.19
53	SM	S–190(5)	849.8	933.4	228.0	232.5	2.92	2.90
54	VAFB	S–188(5)	750.3	928.4	749.9	725.8	98.40	97.80
55	VAFB	S–191(6)	102,006.0	124,476.3	499.1	467.2	90.00	89.79
56	VAFB	S–186(5)	900.3	879.5	230.0	221.0	96.80	97.44
57**	VAFB	S–189(5)	560.1	1174.7	509.8	266.0	97.80	98.04
58	SM	S–187(4)	500.3	569.5	498.3	502.0	2.92	2.86
59	SM	S–194(7)	502.1	522.8	501.9	502.7	2.92	2.99
60	VAFB	S–195(5)	719.5	712.1	358.7	363.0	89.99	90.74
61	VAFB	S–179(4)	1173.7	1068.8	1001.9	993.4	99.88	99.67
62	VAFB	S–197(5)	838.0	792.2	340.3	345.1	90.00	90.32
63	VAFB	S–200(5)	1119.5	1113.6	1102.7	1070.1	90.00	89.92
64	VAFB	S–201(5)	620.6	638.2	619.9	583.4	97.87	97.60
65	WFC	S–202(5)	601.0	657.6	600.0	559.7	54.98	54.94
66	WFC	S–198(5)	626.1	655.9	624.9	607.0	55.00	55.04
67	VAFB	S–203(8)	619.2	578.9	357.0	350.9	97.02	96.97

CONFIGURATIONS:

(1) ALGOL II, CASTOR I, ANTARES II, ALTAIR I
(2) ALGOL II, CASTOR I, ANTARES II, ALTAIR II
(3) ALGOL II, CASTOR II, ANTARES II, ALTAIR II
(4) ALGOL II, CASTOR II, ANTARES II, ALTAIR III
(5) ALGOL IIIA, CASTOR IIA, ANTARES IIA, ALTAIR IIIA
(6) ALGOL IIIA, CASTOR IIA, ANTARES IIB, ALTAIR IIIA, ALCYONE IA
(7) ALGOL IIIA, CASTOR IIA, ANTARES IIB, ALTAIR IIIA
(8) ALGOL IIIA, CASTOR IIA, ANTARES IIIA, ALTAIR IIIA

* FIRST STAGE ANOMALY
** PITCH PROGRAM ANOMALY

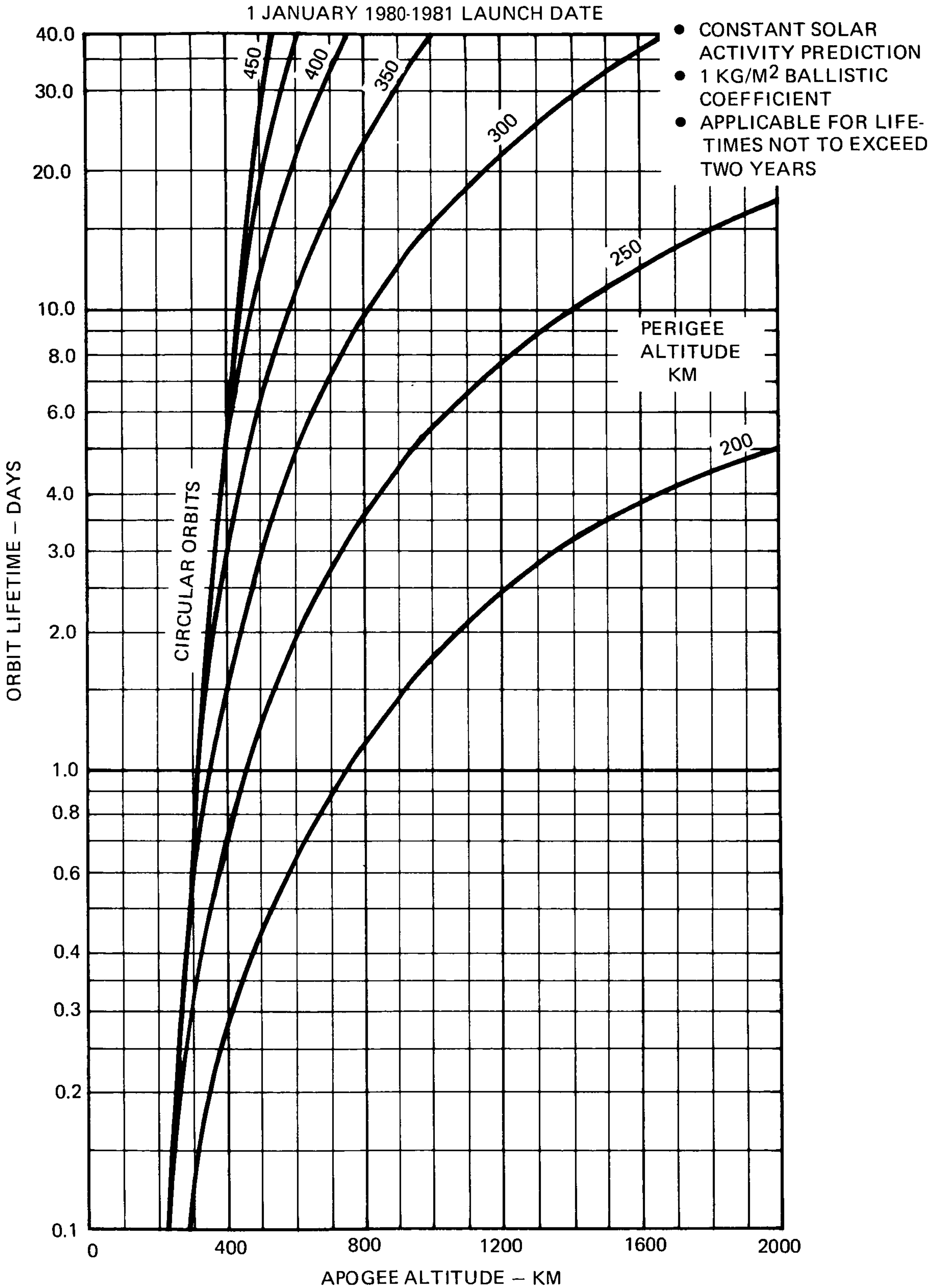


FIGURE II–18 ORBIT LIFETIME – 1980-1981 LAUNCH

3.8 GEO-SYNCHRONOUS TRANSFER ORBITAL PERFORMANCE

An orbit of special interest within the Scout performance spectrum is the geo-synchronous transfer orbit. The Scout vehicle has the capability to perform this mission from the San Marco and Wallops Flight Center launch facilities. Low inclinations can be obtained from San Marco while higher inclinations are possible from Wallops Flight Center.

The geo-synchronous transfer orbit, usually 200 km perigee x 35,800 km apogee, can be obtained with the Scout vehicle by injecting at perigee at fourth stage burnout. If a true geo-synchronous orbit is desired, the energy to raise the perigee from 200 km to 35,800 km and to correct the inclination must be supplied by the spacecraft.

The table below shows the payload weight capability for Scout when launched due east from San Marco and Wallops Flight Center into the geo-synchronous transfer orbit.

LAUNCH SITE	INCLINATION	PERFORMANCE FOUR STAGE 1.07 METER H/S
SAN MARCO	2.9^{o}	63 KG
WALLOPS FLIGHT CENTER	37.7^{o}	58 KG

4.0 PROBE MISSION PERFORMANCE

A Scout probe mission is achieved by imparting the maximum energy of the booster system to the payload, allowing the payload to ascend to a maximum height. The major difference between probe and orbit boost trajectories is in the timing of fourth stage firing. In a probe mission, the fourth stage is nominally fired 21 seconds following third stage burnout instead of after the long third stage coast used for orbit missions. Fourth stage spin up and separation are accomplished during this 21 second period. The spacecraft may or may not be separated from the expended fourth stage following burnout.

The probe mission performance data presented are for Wallops Flight Center launches on an azimuth of 90 degrees. The 90-degree launch azimuth was selected because the maximum performance is achieved at this azimuth.

LAUNCH SITE	INCLINATION	PERFORMANCE FOUR STAGE 1.07 METER H/S	APOGEE ALTITUDE	ZERO "G" TIME
WALLOPS FLIGHT CENTER	37.7°	91 KG	14631 KM	263 MIN.
WALLOPS FLIGHT CENTER	37.7°	181 KG	* 3111 KM	116 MIN.

* An increased apogee altitude can be achieved but only with a decreased zero "g" time.

5.0 RE-ENTRY MISSION PERFORMANCE

The Scout vehicle lends itself well to the re-entry type mission. When employed to fly a re-entry mission trajectory, a significant departure from orbit and probe mission staging sequence is required. The first two stages are handled in the same manner as orbit and probe missions. The third and fourth stages are generally used to drive the payload back into the atmosphere. This necessitates a much longer second stage coast than the minimum five seconds used for orbit and probe missions. This long coast phase allows the vehicle to coast to the vicinity of the apogee altitude resulting from the energy supplied by stages one and two. The exact ignition time and position of the third stage is determined by the re-entry test conditions required by the Payload Agency. Following third stage burnout, a minimum coast phase of 21 seconds is required to perform the spin-up and separation of the fourth stage and payload.

The re-entry performance data presented in the following chart was obtained by simulating gravity turn trajectories. There are a large number of variables available in the design of a re-entry trajectory but the gravity turn method used tends to yield re-entry velocities near the optimum for the vehicle. There are many ways to shape the trajectory to achieve various constraints such as range or entry angle. These, however, usually require vehicle energy in the manipulation of the vehicle's velocity vector which results in a loss in performance.

Typical re-entry performance is presented in the following table for Vandenberg Air Force Base westerly launches. Similar easterly performance is available from Wallops Flight Center. Since payload recovery and interrogation following booster burnout are generally required, WFC is suited for re-entry missions because of the strategic location of Bermuda Island approximately 1100 km downrange.

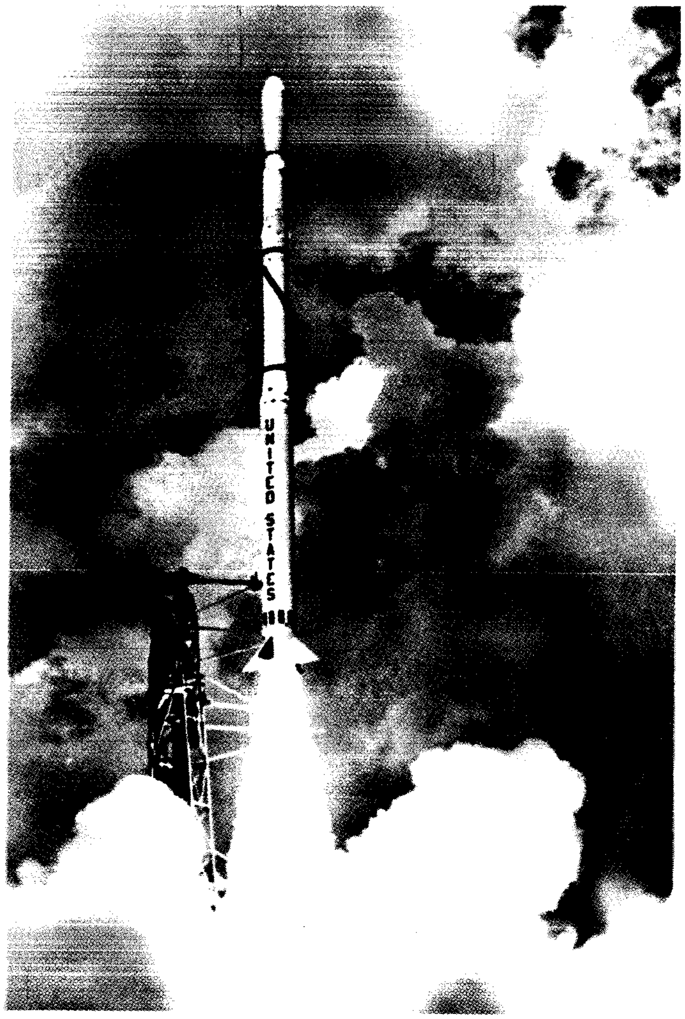
VAFB LAUNCH
PAYLOAD = 270 KG

TRAJECTORY PARAMETERS	NOMINAL VALUE AT RE-ENTRY (FOURTH STAGE BURNOUT)		
RANGE	1200 KM	3700 KM	7780 KM
ALTITUDE	91 KM	122 KM	122 KM
RELATIVE VELOCITY	5.4 KM/SEC	6.1 KM/SEC	6.9 KM/SEC
RELATIVE FLIGHT PATH ANGLE	–20 DEG	–20 DEG	–20 DEG

6.0 SOLAR ORBIT MISSION PERFORMANCE

A solar orbit is achieved by imparting greater than earth escape velocity to the spacecraft. This allows the spacecraft to leave the earth's influence and be captured by the gravitational field of the sun.

Solar orbit missions are achievable with the five stage Scout configuration. The vehicle is flown similar to the earth probe mission but with an additional stage to escape the earth's gravity. Solar orbit performance is dependent upon launch time of day and time of year. Scout performance for a specific mission will be calculated on request.



chapter III
MANAGEMENT AND DOCUMENTATION

SCOUT
user's manual

MANAGEMENT 1

DOCUMENTATION 2

DOCUMENTATION/INTEGRATION MILESTONES 3

1.0
MANAGEMENT

This section describes the Scout overall management organization, identifies the functions and responsibilities of the Mission Working Group, and describes program funding policies.

1.1
SCOUT
MANAGEMENT

The Scout management organization is diagramed in Figure III-1.

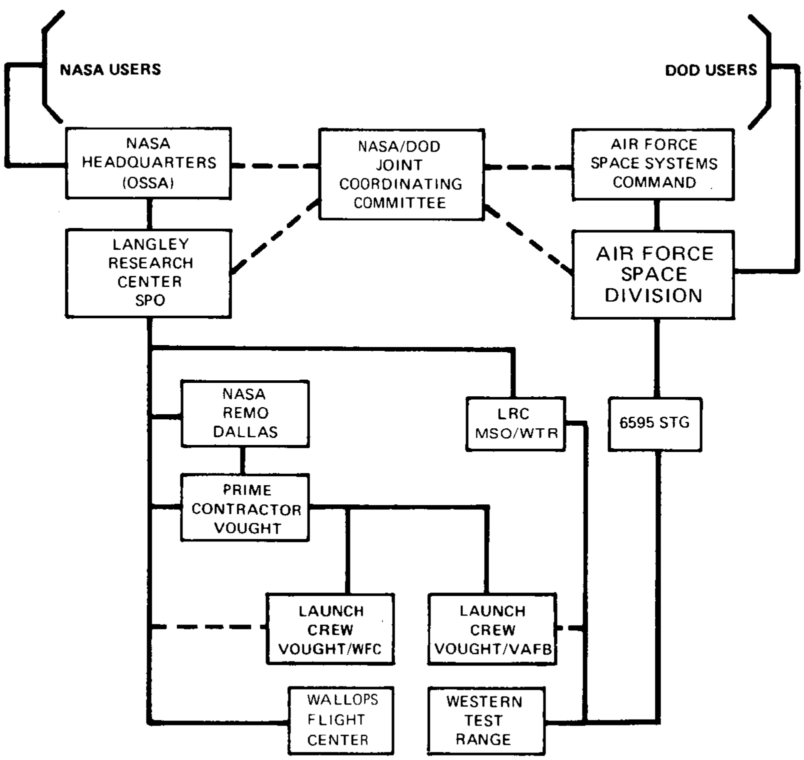


FIGURE III-1 NASA/DOD SCOUT SYSTEM ORGANIZATION

The Scout program is administered through the Medium Launch Vehicles Office of the Launch Vehicles and Propulsion Programs Division of the Office of Space Sciences (OSS) of the National Aeronautics and Space Administration (NASA) headquarters, Washington, D.C. This office provides broad program policy and fiscal management in addition to chairing the Joint National Aeronautics and Space Administration and Department of Defense (NASA-DOD) Scout Coordination Committee. This committee provides the inputs to the Medium Launch Vehicles Office required to integrate both NASA and DOD vehicle requirements into a coherent production and launch program.

Under the direction of NASA Headquarters, the Scout Project Office (SPO) of the Langley Research Center (LRC) is responsible for vehicle systems management maintaining centralized technical direction of procurement, configuration, procedures and checkout.

Vought Corporation, prime contractor to LRC, furnishes services and material necessary to provide system management for the Scout Program. These services include Scout launch teams located at Wallops Flight Center and Vandenberg Air Force Base.

The Deputy for Launch Vehicles, Expendable Launch Vehicles Program Office, Medium Launch Vehicle Operations (LVMM) at Headquarters, Air Force Space Division has prime responsibility for all DOD programs using the Scout booster. Space Division exercises management control of Scout vehicle support for DOD missions through the established Scout system organization. DOD requirements are officially defined to the NASA/DOD Scout Coordinating Committee, NASA Headquarters, and the Scout Project Office as appropriate.

Space Division coordinates overall DOD mission schedules and payload integration to assure timely availability of all elements necessary for accomplishment of specific launch objectives.

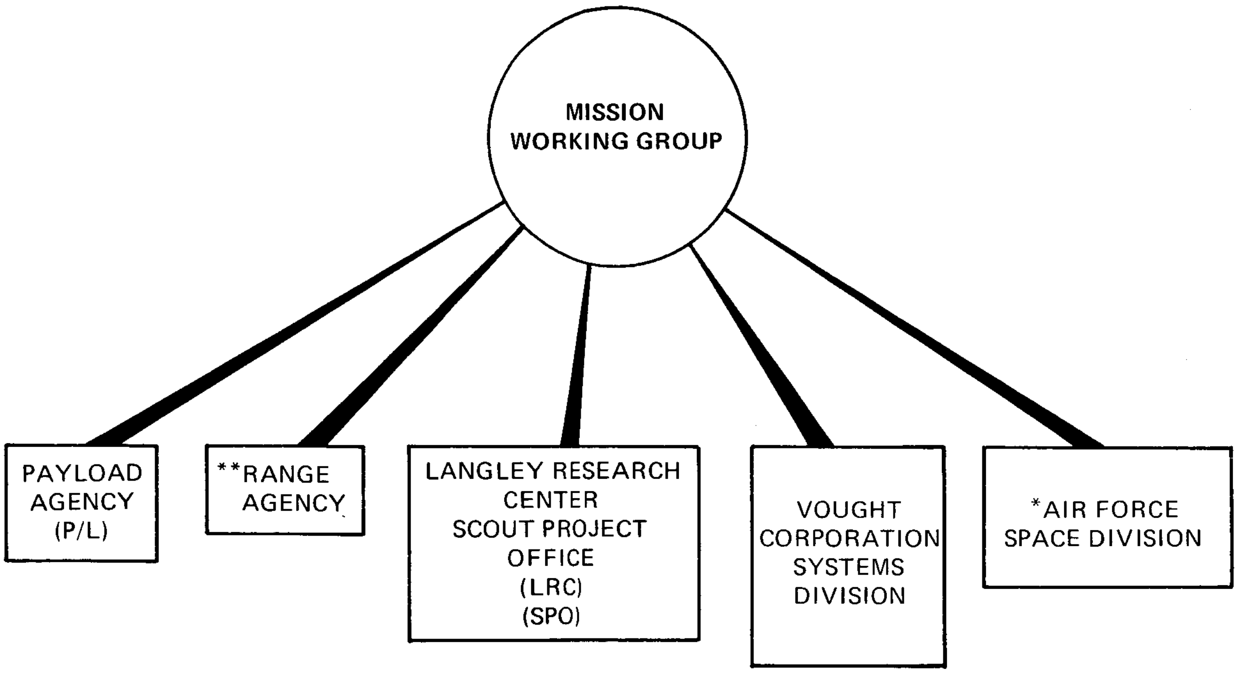
For DOD programs, Space Division is responsible for the timely submission of documentation to the supporting test range. The 6595th Aerospace Test Wing conducts the necessary liaison to assure support of DOD mission requirements. To accomplish this responsibility, Space Division requires payload peculiar information from the program agencies and trajectory and preflight documentation from Vought Corporation.

1.2 PROGRAM MANAGEMENT

Ordinarily, a number of different agencies are involved in mission planning and payload-to-vehicle integration associated with individual programs. To effectively accomplish these tasks, a submanagement organization identified as a Mission Working Group is employed.

The Mission Working Group exercises overall management control of vehicle and range interfaces for each program. Specifically, the group is responsible for the direction of all documentation efforts, the physical and operational integration activities and is, in general, charged with mission responsibility at the working level. The Mission Working Group is formed by LRC or Space Division upon official assignment of a payload to Scout. It is composed of one qualified representative from each prime agency connected with the program. The LRC Scout Project Office and Air Force Space Division Representatives are Co-Chairmen of all DOD payload meetings. The LRC Scout Project Office Representative is Chairman of all other Mission Working Group meetings.

Figure III-2 schematically presents the relationship of the Mission Working Group and its parent agencies.



*FOR ALL DEPARTMENT OF DEFENSE (DOD) VEHICLES

**RANGE REPRESENTATION SHOWN ABOVE IS INVITED AT AN APPROPRIATE TIME PRIOR TO LAUNCH

FIGURE III–2 INTEGRATION PROGRAM MANAGEMENT ORGANIZATION

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The formation of the Mission Working Group does not in any way relieve any agency of any responsibility that is peculiar to that agency. The Mission Working Group makes recommendations peculiar to that particular mission and assures, in effect, that these actions are indeed carried out in a timely manner by the responsible agencies. Full responsibility for these functions is assumed by the Mission Working Group Chairman.

Payload Agency contact with the prime contractor is recognized as vital and is unlimited if made through the proper channels and in the proper context. The Scout User should understand his position with respect to the Scout vehicle prime contractor, Vought Corporation, and the Langley Scout Project Office. Only the proper authorities at LRC may direct the prime contractor to make changes with respect to any of the Scout contracts. For contractual reasons, the proper channel for change requests, proposal requests, actions by Vought Corporation or actions for the Langley Scout Project Office is through the Mission Working Group. In the interest of expediting, such contacts may be made verbally; however, all agreements should be immediately confirmed in writing in order to preclude any misunderstanding.

Experience has shown that the optimum lead time for integrating program activities is 18 months. This allows ample time but not excessive time from vehicle assignment to launch.

1.3 PROGRAM FUNDING

The purchase price of a Scout vehicle (payload excluded) includes basically all hardware, rocket motors and vehicle systems along with all associated launch services required to inject a standard four or five stage Scout into a probe, re-entry, orbital or escape trajectory. In addition, a Mission Feasibility Study, a Preliminary Trajectory Analysis, and a Final Trajectory Analysis are provided the Scout user on a "one time only" basis. This software is more fully described in a subsequent section on Documentation.

For extra services or extra standard Scout hardware, or standard Scout design information, the Langley Scout Project Office will act as the procurement agency for the requisitioning agency.

In cases where it is necessary to change any part of the standard Scout system through additional parts procurement or design change to better suit the payload requirements, the change must be authorized by the Langley Scout Project Office. The SPO will work directly with the Payload Agency or through Space Division for DOD payloads to determine the extent of the changes required and with Vought to determine the effect of the changes in the Scout system. Actual fund transfer arrangements must be made with the Scout Program Manager at the Office of Space Services

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at NASA Headquarters, Washington, D. C. The Langley Scout Administrative Officer will make the preliminary arrangements with OSS, and thereafter OSS and the Payload Agency will deal directly with each other furnishing Langley Scout Project Office an information copy of all correspondence.

Space Division exercises fiscal management control for DOD programs and will procure vehicle support for the respective program.

The Range services required for vehicle launch, tracking and data acquisition are provided by the Range agency. These services are normally provided by ranges in the United States at no direct cost. The Range User's Manual for a particular range should be consulted for the services available.

2.0 DOCUMENTATION

Documentation is required to provide the information necessary for support of a program which must pass between the contributing agencies to achieve a timely and successful program. The purposes of the payload documents are as follows:

- To answer questions relative to the payload which will be asked by other agencies and/or contractors involved.
- To aid in program management of the integration process.
- To describe payload interfaces, both physically and electro-mechanically, so that the launch vehicle and payload may be integrated in a timely and precise manner.
- To identify personnel who are responsible for the payload integration and operation.
- To define range and facility requirements.

In general, a Scout User will be required to prepare some of the documents in their entirety; he will provide inputs to others and will receive documents prepared by other agencies in response to his documents. The ensuing paragraphs identify the documents used for planning and integration and describe their purpose, scope and responsibility for preparation.

For ease of discussion, the documents have been grouped in categories of Mission Planning Documents, Range Documents, and Hardware and Operational Interface Documents.

2.1 MISSION PLANNING DOCUMENTS

Mission Planning documentation requires inputs from the Payload Agency very early in the program. The documented analyses assist the user of Scout to define the final mission requirements that are consistent with the vehicle performance capability.

2.1.1 Mission Feasibility Study

A Mission Feasibility Study is prepared by Vought Corporation for those missions not covered in the published performance curves. Inputs are required from the Payload Agency through the Mission Working Group. The study determines the most accurate and acceptable type preliminary trajectory that will satisfy the mission requirements. If necessary, this may be in the form of a series of small studies; e.g., determine the maximum payload weight to use for a specific orbit.

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2.1.2 Trajectory Analysis

A Preliminary Trajectory Analysis is prepared by Vought when the spacecraft weight and mission requirements become firm. Inputs are required from the Payload Agency through the Mission Working Group. Data requirements consist of payload definition, mission requirements, launch site selection, payload environmental restrictions, and motor performance. The analysis document will provide a complete nominal ballistic trajectory, radar look angles, and an orbital ephemeris for use in preliminary planning. Inputs are required nine months prior to launch.

The Final Trajectory Analysis is prepared by Vought when the motor and vehicle assignment has been made. Inputs are required from the Payload Agency through the Mission Working Group. Data requirements are the same as the preliminary analysis except that all data is firm and final. The output of the analysis provides a Trajectory Data Package which includes a predicted boost trajectory to fourth stage burnout, expended stage impact trajectories, ephemeris of predicted orbit, instantaneous impact points and radar look angles, as required. Final data must be available no later than 60 days before launch. The Trajectory Data Package is available 30 days before launch.

2.1.3 Trajectory Data Package

The Trajectory Data Package is prepared by Vought and is the output of the trajectory analyses. Firm data for the final trajectory analyses must be available no later than 60 days before launch. The Trajectory Data Package is available 30 days before launch and includes a predicted boost trajectory to fourth stage burnout, expended impact trajectories, ephemeris of predicted orbit, instantaneous impact points and radar look angles, as required.

2.1.4 Pre-flight Brochure

The Pre-Flight Brochure is published by Vought approximately thirty days prior to launch and contains technical information pertinent to the particular vehicle. The data presented includes such items as the vehicle commanded pitch program, detailed tabulations of computed trajectory parameters, moments of inertia information, radar look angles, detailed vehicle weight summary by components and major subassemblies, and predicted impact dispersion areas for the various stages (and payload when applicable). Inputs are required 60 days before launch.

2.1.5 Final Flight Report

A Final Flight Report is published by Vought sixty calendar days after available raw data is received. A Flash Flight Report is issued by the Launch Team the day following the launch and includes a description of the objectives and how they were met, conditions of the vehicle at launch and a summary of flight trajectory and vehicle performance based on a quick look at radar and telemetry data.

Raw data obtained from each Scout flight is forwarded to Vought for analysis and documentation in the Final Flight Report. This report contains a detailed engineering analysis and evaluation of the vehicle performance and its systems. Special emphasis is placed on any investigation or special

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tests performed as a result of a problem or unusual occurrence during the flight. The spacecraft analysis and evaluation is the responsibility of the Payload Agency and/or contractor and is usually not discussed in this report.

2.2 RANGE DOCUMENTS

Range documentation requirements vary between ranges. This variance is primarily due to the Western Test Range being a National Range, Wallops Flight Center a NASA Service Range, and the San Marco Equatorial Range an Italian-owned range. However, from the Scout user's standpoint, each range imposes similar requirements.

2.2.1 Air Force Western Test Range Documents

The documentation requirements at the Space Missile Test Center's (SAMTEC) Western Test Range are governed by the documentation system established for National Ranges and are fully described in the Range User's Handbook (SAMTEC 80-1).

The NASA Field Project Office (FPO/WTR) at VAFB will coordinate spacecraft information and requirements for NASA spacecraft.

The necessary booster information, range safety information, and other operations support requirements for launch, are prepared by Vought Systems and submitted to SAMSO for DOD programs.

In addition to the National Range documentation, the Scout Vehicle System has the requirement for one additional document, the Payload Description Document (PDD).

2.2.2 Wallops Flight Center Documents

Normally, the Payload Description Document is sufficient for most programs.

2.2.3 San Marco Equatorial Range Documents

The documentation requirements at the San Marco Equatorial Range are described in the San Marco Range User's Manual. Two documents are normally required; the Program Introduction and the Payload Description Document.

2.2.4 Program Introduction (PI)

The Program Introduction Document is required for launches from VAFB and San Marco. The PI informs the range that a program is being planned and outlines the support that will be required from the range.

2.2.5 Program Requirements Document (PRD)

The Program Requirements Document is required for launches from VAFB.

The PRD further defines the range support needed by the Range user. The PRD is a series of standard form pages, each pertaining to a particular requirement, and is prepared by the Range user.

The Range responds to the PRD with a Program Support Plan (PSP) describing the methods by which the range intends to provide the requested support.

2.2.6 Operations Requirements (OR)

The Operations Requirements Document is required for launches from VAFB.

The OR is prepared by the Range user and describes in detail the specific items of an operational nature required from the range by the program. For major programs, the OR supplements the PRD and confirms the general requirements as stated in the PRD and committed in the PSP.

The Range responds to the OR with an Operations Directive (OD) which describes in detail the operational support being provided by the range.

2.2.7 Payload Description Document (PDD)

The Payload Description Document is required for all Scout launches.

The Payload Description Document is similar to the PRD; differing primarily in format. The PDD describes the support requirements of the Range user's program. The format for a PDD may be found at the end of this section as well as in the Wallops Flight Center User's Handbook and the San Marco Range User's Manual. The format shown should be used as a guide and may be tailored to meet the needs of the particular program and the requirements of the supporting range.

Distribution of the Payload Description Document will be as follows:

Wallops Flight Center (WFC launches)	20 copies
VAFB (VAFB launches)	5 copies
LRC/FPO/WTR (VAFB launches)	10 copies
San Marco Range (San Marco launches)	5 copies
NASA LRC (all launches)	10 copies
Vought (all launches)	5 copies

The PDD is prepared by the Payload Agency and is submitted to the Mission Working Group at least six months prior to the planned mission.

2.2.8 Operations Plan

The Operations Plan is used as the prime information source by all agencies concerned with the particular operation. The Operation Plan is used primarily at Wallops Flight Center and San Marco. The Payload Agency will be required to provide inputs to this plan which is issued 21 days prior to launch by the Vought launch team.

2.2.9 Range Safety (Flight and Ground)

The Range Safety documentation required at VAFB is prescribed in the WTR Range Safety Manual, SAMTECM 127-1. The Payload Agency will be required to provide inputs to the safety document through the Mission Working Group. Mission requirements and payload characteristics must be available no later than four months prior to launch. Safety documentation required by Wallops Flight Center and San Marco is prescribed in the Range User's Manual for these ranges.

2.2.10 Support Instrumentation Requirements Document (SIRD)

The Support Instrumentation Requirements Document is required by NASA for missions which a Payload Agency desires instrumentation support from the NASA tracking, data acquisition, and communication systems. The SIRD is a series of standard form pages, each pertaining to a particular requirement, and is prepared by the Payload Agency through the Mission Working Group. The SIRD is required 12 months prior to launch.

2.3 HARDWARE AND OPERATIONS INTERFACE DOCUMENTS

The electro-mechanical interface of the payload to the Scout system is documented on Vehicle and Ground Support Equipment Interface Drawings. The operational interface is controlled by the Countdown Manual. Vought Corporation as System Manager of the Scout Program, is responsible for the preparation of these documents. Inputs will be required of the Payload Agency although much of the information is included in the Payload Description Document.

2.3.1 Payload Interface Drawing – Vehicle

This drawing is prepared by Vought from information furnished by the Payload Agency. When approved by the Mission Working Group members, the drawing establishes formal identification of all payload/vehicle technical agreements. Information is required approximately six months prior to launch.

A typical Payload Interface Drawing – Vehicle is shown in Figure III-3.

This drawing will also be used by the launch crews to process the fourth stage of the vehicle in the area of payload/vehicle interface.

2.3.2 Payload Interface Drawing – Ground Support Equipment

This drawing is prepared by Vought in the same manner as the Vehicle Interface Drawing and defines the payload interface with the Scout Launch Complex. Such items as blockhouse console space allocation, payload environmental provisions, location of special payload checkout vans or equipment, and payload umbilical wiring are covered in this drawing. The drawing is issued in final form approximately 90 days prior to launch. No work is accomplished on the launch complex to meet the mission requirements until this drawing is approved by the Mission Working Group and released. Inputs from the Payload Agency should be available six months prior to launch.

A typical Payload Interface Drawing – Ground Support Equipment is shown in Figure III-4.

2.3.3 Countdown

The vehicle countdown is prepared by Vought Corporation at Wallops Flight Center and VAFB and by the CRA at the San Marco Range. The countdown prescribes the launch operation sequence to satisfactorily accomplish all prelaunch and countdown activities that will culminate in a successful launch. Inputs are required of the Payload Agency to achieve an integrated countdown. The countdown preparation is completed 30 days prior to launch.

FORMAT
FOR
PAYLOAD DESCRIPTION DOCUMENT

1.0 ADMINISTRATIVE DATA

1.1 PRIME GOVERNMENT AGENCY

State the agency, division, branch, section, etc.

1.2 SUPPORT AGENCIES

All agencies supporting the mission should be listed.

1.3 PRIME PAYLOAD CONTRACTOR

State the name and address of the prime contractor.

1.4 OFFICIAL NASA PROJECT DESIGNATION

State the official NASA project designation.

1.5 OTHER DESIGNATIONS

Other project designations "nicknames," prior nomenclature, etc., should be listed.

1.6 RESPONSIBLE PROJECT PERSONNEL

Give the title, responsibility, name, business and home addresses, and telephone numbers of all responsible project officers.

1.7 CLASSIFICATION LEVELS

List all classified phases of the program and the degree of classification.

2.0 GENERAL INFORMATION

2.1 TEST DESCRIPTION

2.1.1 PURPOSE OF THE TEST

List the primary and secondary objectives of the test.

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2.1.2 LAUNCH WINDOW

Give the preferred launch time and the allowable period. Also state the controlling factors for determination of the launch window and who may authorize deviation.

2.1.3 MISSION DESCRIPTION

State briefly the purpose of the mission and the objectives to be met. In addition, the desired mission constraints and maximum deviations from these constraints allowable for mission success will be specified as follows:

Orbital Mission

Perigee	________	±	________
Apogee	________	±	________
Inclination	________	±	________

Reentry Mission (at reentry)

Altitude	________	±	________
Relative velocity	________	±	________
Flight path angle	________	±	________
Flight azimuth	________	±	________
Impact area (coordinates)	________	±	________

Probe Mission

Peak altitude	________	±	________
Flight azimuth	________	±	________
Duration of zero "g"	________	±	________

Other constraints if applicable

Final definition of the planned trajectory will be accomplished by the Scout Project Office. Copies of the final trajectory print-out will be furnished to the payload agency.

2.1.4 RECOVERY

State and describe what is to be recovered, reasons for recovery, hazards involved, and any recovery aids and their characteristics. Include aids such as chaff (frequency, quantity), SARAH beacon (frequency, power output, period of operation), dye marker (color, persistence, time of deployment), flashing light (color, frequency, duration, candle power, directional characteristics), smoke (color, duration, time of deployment), radar reflective parachute (when deployed, size), or any other aids used. Also give the desired period of recovery operations and the disposition of the recovered item.

2.2 PAYLOAD LAUNCH RESTRAINTS

2.2.1 WEATHER

State local and downrange weather restrictions other than those pertaining to vehicle design limitations and range operational constraints. State the maximum cloud cover allowable and the necessary visibility. Give any other limiting conditions such as precipitation, humidity, temperature, etc.

2.2.2 TECHNICAL DIFFICULTIES

State conditions which will necessitate cancellation or a mandatory hold, exclusive of range and vehicle operation requirements, due to lack of support from the following areas: communications, payload telemetry, ground telemetry, mobile telemetry, ground radar, mobile radar, or optical coverage.

2.2.3 REVIEW OF PAYLOAD LAUNCH RESTRAINTS

State who is responsible for review of payload launch restraints to determine possible relaxation of test conditions and who shall have final authority on relaxing any of the payload test requirements.

3.0 PAYLOAD TECHNICAL DATA

3.1 PAYLOAD DESCRIPTION

The purpose of the payload description is to identify the interface

requirements for mating the payload to the Scout launch vehicle. Where drawings and/or schematics are specified herein, the requirements may be fulfilled without consideration for multiple distribution by including the required drawings and/or schematics as an integral part of the Payload Description Document. Provided this means of distribution is selected, all dimensional data must be legible.

3.1.1 PAYLOAD DIMENSIONS

Payload drawings should be supplied showing all dimensions with particular emphasis on heat shield, umbilical and upper-stage interfaces. One copy of each drawing will be forwarded to the applicable Range, SPO and Vought.

3.1.2 PAYLOAD WEIGHT BREAKDOWN

Payload weight should be tabulated by major components, including structural instrumentation and total weight.

3.1.3 MOMENTS OF INERTIA

Payload moments of inertia should be tabulated by major components along with the total moments of inertia.

3.1.4 PAYLOAD DYNAMICS

To insure that the PAYLOAD is placed and positioned correctly in the desired orbit, information concerning the PAYLOAD dynamics should be submitted. This information will be used in a pre-launch dynamic analysis of the 4th stage during burn and coast. Such data as, attitude, time history of coning and spin rate, and stability characteristics of the PAYLOAD will be determined in this analysis. The information required for this type of dynamic analysis, in addition to weight, C.G. location and moments of inertia, is a description of the PAYLOAD'S systems of possible energy dissipation, such as nutation or precession dampers, tanks of fluid, solar paddles, antenna systems, experiment booms, etc.

3.2 HAZARDOUS MATERIALS

3.2.1 PYROTECHNIC DETAILS

Seven readily distinguishable copies of schematics and

wiring diagrams of all pyrotechnic circuits and all other circuits physically or electrically related to pyrotechnics are required. Five copies will be forwarded to the Range, one copy to the SPO and one copy will be forwarded to Vought. For each squib, show on the drawings the minimum sure-fire current, maximum no-fire current, recommended firing current, nominal resistance and, if available, the r-f characteristics.

Give a description of the power source, including output, battery life, and details on battery charging. Scale drawings must be supplied for any payload having r-f transmitters or beacons showing the location of all pyrotechnic devices in relation to all transmitting antennas. The frequency, range, type of emission, type of radiating antenna and radiated power (both peak and average) shall be shown for each transmitter or beacon. Schematics, drawings, and operational description of pyrotechnic checkout and monitoring equipment, and any other auxiliary equipment also must supplied. The Range must be notified of any changes as it is the range user's responsibility to certify that all drawings are up to date. All pyrotechnic details must be approved by the Range.

3.2.2 RADIOACTIVE MATERIALS.

If radioactive materials are used in the payload or any of the checkout procedures, the information in paragraphs 3.2.2.1 and 3.2.2.2 must be supplied.

3.2.2.1 RESPONSIBILITY FOR RADIOACTIVE MATERIALS

a) For all radioactive materials planned for use at this Station which will involve exposure or possible exposure of personnel, application will be made by Wallops Flight Center to the Atomic Energy Commission for a license granting this Station the authority to:

1. Handle, store, ship, and control sources in use at this Station.

2. Establish operational procedures and provide monitoring and the required records.

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3. Establish necessary emergency procedures in the event of malfunctions, explosions, or destruct action.

4. Dispose of waste materials.

b) Permission may be granted by Wallops Flight Center for a licensed range user to possess and control small calibration or other small sources at this Station providing:

1. An operational procedure is submitted for storage, handling, shipment, etc.

2. Records are maintained of all radiation sources, etc.

3. He accepts responsibility for the source as stated in his license.

3.2.2.2 TECHNICAL INFORMATION ABOUT RADIOACTIVE MATERIALS

The following technical information also must be submitted:

a) Types and numbers of radioactive materials with their current curie content.

b) Size, shape, and general characteristics of the radioactive sources.

c) Mission of each source.

d) Radiation level versus distance from material.

e) Container descriptions.

f) Shipping and storage container and label descriptions.

g) Shipping date and method of shipping.

h) Two copies of the range user's application for license and two copies of Atomic Energy Commission's license details.

i) Range user's personnel monitoring devices and methods of use (portable survey instruments, personnel dosimeters, film badges, procedures, etc.).

j) Location of radioactive source on research vehicle.

k) Person (range user) at the Range who shall have responsibility.

l) Prior to operations at this Station, a record of exposure of each individual who will be exposed at the Range. This should include AEC - Form 4 along with total exposure, last exposure date, etc.

m) A detailed breakdown of estimated time of source exposure during all buildup, test, and launch operations.

n) Procedure for handling and use of external sources during all times exposed.

o) All calibration procedures involving the use of exposed radioactive sources.

3.2.3 CHEMICALS

Describe all chemical systems, including toxicity and necessary precautions to be taken.

3.2.4 PRESSURE VESSELS

If pressure vessels are used in the payload, supply all technical characteristics, including details on design pressure and test pressure.

3.2.5 OTHER HAZARDOUS MATERIALS

Give technical details and precautions for any other hazardous materials included in payload.

3.3 OTHER ELECTRICAL DETAILS

3.3.1 PAYLOAD SCHEMATICS AND CIRCUIT DRAWING

3.3.1.1 TELEMETRY AND EXPERIMENT DIAGRAMS

Give telemetry and experiment block diagrams. Circuit drawings are required if the telemetry or experiment circuits are related to pyrotechnics.

3.3.1.2 UMBILICAL CIRCUITS

Connector wiring should be tabulated by pin assignment, function, wire size, connector description and location.

3.3.2 PAYLOAD TELEMETRY SYSTEM

A complete description of the telemetry system, its operation and its instrument sensors should be provided. To be included is the method of providing timing signal, power source and output, battery life, battery charging, handling and replacement, accessibility, prelaunch operation of system, system environment restrictions, and effects of payload operation on booster systems environment.

3.3.3 TRANSPONDER AND/OR BEACON SYSTEM

Give the transmitter and receiver frequencies along with the transponder delay time. If a CW beacon system, give the beacon frequency.

3.3.4 COMMAND SYSTEM

Give all electrical details and operational characteristics of any payload command system.

3.3.5 OTHER ELECTRICAL SYSTEMS

Give all electrical details and operational characteristics of any other payload systems.

3.4 FLIGHT SEQUENCE OF PAYLOAD

Give a complete activation and flight sequence of events.

3.5 SEPARATION SYSTEM

Schematics and assembly drawings should be used to show the detail and operation of the separation system. One copy of each drawing will be forwarded to the Range, SPO and Vought. In the case of a Scout furnished separation system, the applicable drawings will be provided by the SPO.

3.6 HEAT SHIELD DETAILS

Heat shield details should be listed, including drawings with overall dimensions, forward tip station, bumper forward face station, access hatch requirements, and umbilical door location and size. If heat shield is supplied by the payload agency, design parameters, details on environment testing, and details on heat shield separation should be supplied. In any case, the final heat shield drawings will be furnished by the SPO.

3.7 SPIN RATE

The required spin rate should be noted, as well as acceptable variations and maximum limitations of spin rate.

3.8 PAYLOAD ENVIRONMENTAL DATA

State any environmental restraints.

3.9 RANGE USER'S GROUND SUPPORT EQUIPMENT

State in detail the ground support equipment which will be supplied by the range user. Also, describe the electromagnetic radiation characteristics of special equipment, including the purpose or function of the equipment, type of emission, average and peak power output, modulation characteristics, antenna description, and expected location of equipment with respect to the test vehicle launch site.

4.0 RANGE SUPPORT REQUIREMENTS

4.1 RANGE INSTRUMENTATION REQUIREMENTS

4.1.1 WALLOPS SUPPORT

4.1.1.1 RADAR

List trajectory tracking requirements and any special calibration requirements, mode of operation (skin or beacon track), frequency and power of beacon, beacon delay time, data period, and data requirements.

4.1.1.2 TELEMETRY

For telemetry support, describe the period of operation, type of output desired, and data handling requirement. Also give in tabular form the number of records and, for each record, give the frequency, modulation, data time period, number of channels, and paper speed. For each channel, give the filter Hz , channel assignment subcarrier KHz , commutation channels, range capability of sensor, and full-scale deflection. The following tabular form should be used.

RECORD NO. X
FREQUENCY XXX
MODULATION XXX
DATA TIME PERIOD

CHANNEL	FILTERS	CHANNEL ASSIGNMENT	SCO	TOTAL DEFLECTION	RANGE
X	XX	XXXX	XX	XX	XX
X	XX	XXXX	XX	XX	XX

Indicate special equipment required from the Range over and above normal telemetry support, such as receivers, recorders, telemetry racks, antennas, decommutables, etc.

4.1.1.3 PHOTOGRAPHIC COVERAGE

Give the type of photographic support needed, such as still and/or motion pictures, including buildup documentary and tracking film. Specify black and white or color and quantity. Documentary coverage requires complete script at least 15 days prior to any photo coverage.

4.1.1.4 COMMUNICATIONS

Describe communication requirements including radio, intercom, and telephone.

4.1.1.5 METEOROLOGICAL DATA

State any special meteorological support requirements, such as sounding rockets, with a schedule and justification for the support.

4.1.1.6 RADAR DATA REDUCTION REQUIREMENTS

State the type of data needed, such as raw, raw-positional, or smooth-positional data, along with the data period and printout interval. Also state whether individual data from each radar or the best available data are required.

4.1.1.7 COMMAND FUNCTION REQUIREMENTS

Give a complete description of the command function requirements.

4.1.1.8 TIMING REQUIREMENTS

Give type of timing code required.

4.1.1.9 RANGE PROGRAMMER REQUIREMENTS

State any functions which should be performed by the Range programmer.

4.1.1.10 CLOSED CIRCUIT TV

State any closed circuit TV requirements.

4.1.1.11 LOCAL R-F FREQUENCIES

State all local r-f frequencies required for the range user's equipment and their use.

4.1.1.12 ADDITIONAL REQUIREMENTS

Describe any additional instrumentation requirements.

4.1.2.7 METEOROLOGICAL DATA

State any special meteorological support requirements, such as sounding rockets, along with a schedule and justification for the support.

4.1.2.8 DATA REDUCTION REQUIREMENTS

State the radar data reduction required in accordance with Paragraph 4.1.1.6.

4.1.2.9 ADDITIONAL REQUIREMENTS

Describe any additional downrange requirements.

4.1.3 LOOK ANGLES

State all look angle requirements

4.2 DATA DISPOSITION

Tabulate the data required from all sources, the number of copies needed, and to whom the data should be sent. Use the following tabular form:

ITEM NO.	SHORT DESCRIPTION	NO. OF COPIES	WHEN REQUIRED	RECIPIENT
X	XXXXXXXXX	XX	XXX	XXXXXX

4.3 PAYLOAD PREPARATION SUPPORT

4.3.1 RANGE FACILITIES

State requirements for range facilities, including work shop, machine shop, etc.

4.3.2 DYNAMIC BALANCE REQUIREMENTS

For dynamic balancing, the range user is required to furnish the following: center of gravity, dynamic balancing speed, any special environmental conditions such as temperature, humidity, maximum acceleration and deceleration, and any special balance procedures.

4.3.3 INERTIA DETERMINATION

State the requirements for experimentally determining moments of inertia of the payload. Also describe the methods to be used and the attach points for swinging.

4.3.4 GROUND SUPPORT EQUIPMENT REQUIRED

State the ground support equipment required from the range in detail. Consideration should be given to relative location, electrical and mechanical assembly areas and furnishings, heavy and light equipment, and machinery and transportation facilities.

4.3.5 PAYLOAD ENVIRONMENT

State any environmental contraints.

4.3.6 R-F SUPPORT REQUIREMENTS

State any r-f support requirements, including any shielded rooms needed for payload checkout.

4.4 BLOCKHOUSE REQUIREMENTS

Blockhouse requirements, such as space, blockhouse to payload wiring, power, battery charging, etc., should be stated.

4.5 PHYSICAL FACILITIES

Requirements for offices, shops, or other work areas should be given.

4.6 ADDITIONAL SUPPORT REQUIREMENTS

4.6.1 RANGE ENGINEERING SERVICES

State any engineering services required.

4.6.2 TECHNICAL SERVICES

State any technical services required.

4.6.3 ADMINISTRATIVE MANAGEMENT SERVICES

State any administrative management services required.

5.0 OPERATIONAL

5.1 PERSONNEL LISTING

Responsible project personnel at the launch site and downrange facilities should be listed by name, function, and location. An estimate of the number of official observers at each facility should be made.

5.2 DELIVERY SCHEDULE

Give a delivery schedule, including dates of arrival and a parts list of all payload hardware.

5.3 OPERATION PLAN

A generalized schedule of tentative milestones should be given, commencing with occupation of launch site and downrange facilities. All preflight checks requiring Range or vehicle support should be included. Payload countdown information is required for integration with the vehicle and range countdown procedures.

3.0 DOCUMENTATION/INTEGRATION MILESTONES

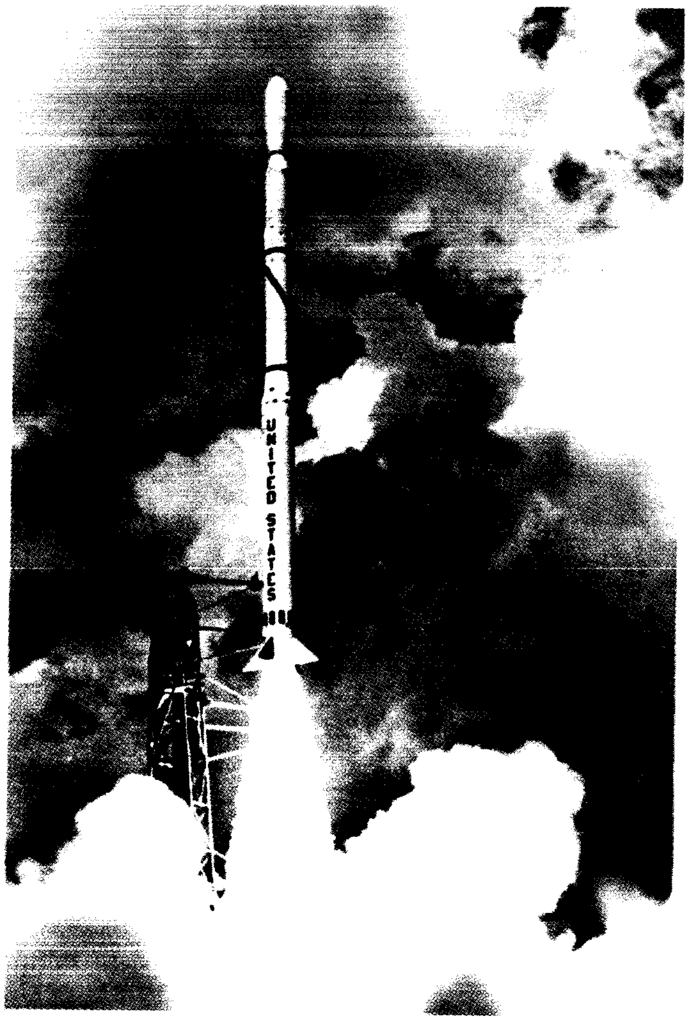
Listed below is a summary of the major documentation/integration milestones for programs utilizing the Scout launch vehicle. It assumes that a program is assigned to Scout twenty-four months prior to launch. The general sequence of events and the events themselves should be adhered to regardless of the time scale for the program.

Detail scheduling for each program is established and coordinated through the Mission Working Group.

Launch Time Minus	Event (Milestones)	Agency Responsible
24 Months	Assign Payload Agency Program to Scout	NASA Hdq
	Form Mission Working Group	LRC-Space Division
	Conduct Mission Feasibility Studies	Vought
	Select Launch Site	MWG
	Provide Design Data to Payload Agency	Vought
	Identify Documentation Requirements	
12 Months	Mission Working Group Meeting	LRC-Space Division
	Establish Estimated Launch Date	MWG
	Determine Performance and Accuracy Requirements	Payload Agency/ Vought
	Define Payload Configuration	Payload Agency
	Establish Heatshield Preliminary Design Parameters	Vought
	Define Fourth Stage Spin Requirements	Payload Agency/ Vought
	Submit Range Documentation	Payload Agency/ Space Division/ Vought
	Submit Support Instrumentation Requirements Document	Payload Agency
9 Months	Mission Working Group Meeting	LRC-Space Division
	Establish Finalized Launch Date	MWG
	Identify Booster Trajectory Constraint	Vought
	Define Preliminary Mission Requirements	Payload Agency
	Provide Preliminary Trajectory Data	Vought
	Complete Heatshield Definition	Vought

Launch Time Minus	Event (Milestones)	Agency Responsible
7 Months	Provide Layout of Payload and Heatshield	Vought
	Assign Scout Vehicle	SPO/LRC
6 Months	Mission Working Group Meeting	LRC-Space Division
	Submit Payload Description Document	Payload-Agency
	Review Range Documentation Requirements	MWG
	Define Heatshield Test Requirements	Vought/ Payload Agency
	Define Wiring Requirements for Payload Umbilical Cabling	Payload Agency
5 Months	Nominal Mission and Motor Data Available	Vought/ Payload Agency
	Prepare Finalized Range Safety Data	Payload Agency/ Vought/Space Division
3 Months (Field)	Mission Working Group Meeting	LRC-Space Division
	Finalize Date for Payload Arrival at Launch Site	MWG
	Release Final Payload Interface Drawing – GSE	Vought
	Perform Heatshield Fit and RFI Test at Dallas	Payload Agency/ Vought
	Provide Finalized Data for Trajectory Analysis and Pre-Flight Planning Report	Payload Agency/ Vought
T – 60 Days	Release Final Payload Interface Drawing – Vehicle	Vought
T – 30 Days	Release Pre-Flight Planning Report	Vought
	Release Final Trajectory Data Package	Vought

chapter IV
FACILITIES AND OPERATIONS



SCOUT user's manual

1.0 RANGE FACILITIES

Presently there are three locations with the facilities capable of launching the Scout vehicle. Two of these, Wallops Flight Center and Vandenberg Air Force Base are located in the United States. The third one, San Marco, is located off the east coast of Africa. This chapter describes the Scout launch facilities at these launch sites. (See Figure IV-1).

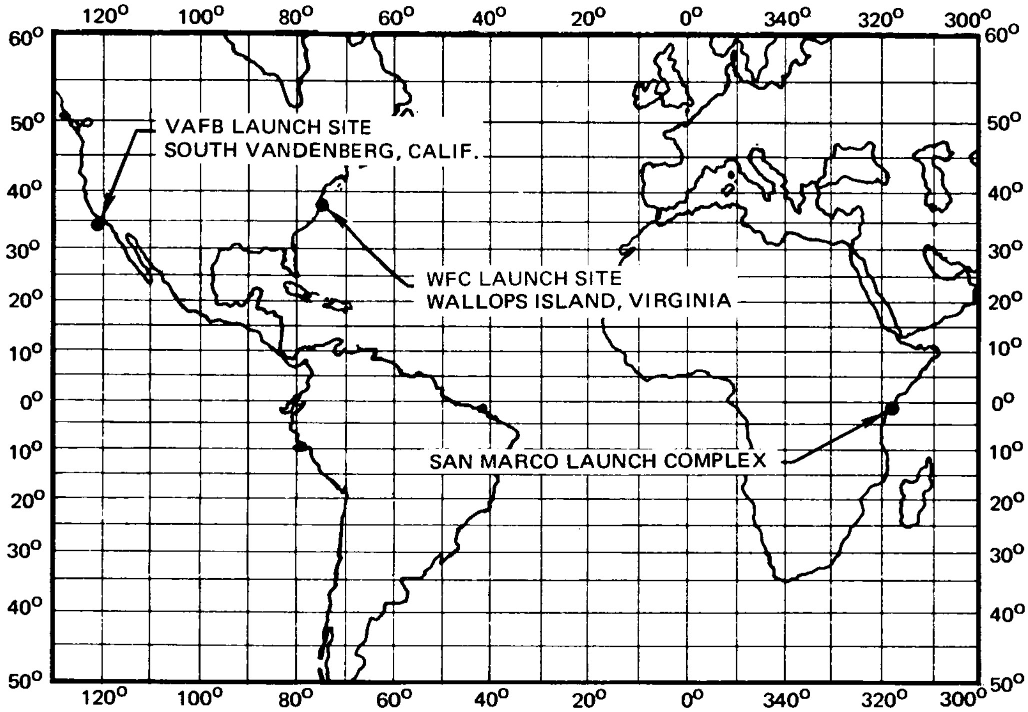


FIGURE IV–1 SCOUT LAUNCH FACILITY LOCATIONS

Additional information concerning the support services and other facilities available at these launch sites may be found in the Range User's Manuals published by each range. These manuals may be obtained from the following addressees:

Publication	Obtained From
Wallops Management Instruction (WMI 1771.1)	National Aeronautics and Space Administration Wallops Flight Center Wallops Island, Virginia 23337
Range User's Safety Handbook (WHB 1771.1)	
Range User's Handbook SAMTECM 80-1	Directorate of Plans (XP) SAMTEC Vandenberg AFB California 93437
San Marco Range User's Manual	National Aeronautical and Space Administration Langley Research Center Scout Project Office Hampton, Virginia 23365

1.1 WALLOPS FLIGHT CENTER (WFC)

The launch site for Scout vehicles on the East Coast of the United States is located at the Wallops Flight Center facility on the Atlantic Coast of the Delmarva Peninsula. (See Figure IV-2).

Wallops Flight Center consists of the Wallops Island firing range and the administrative and technical service support facilities on the nearby mainland.

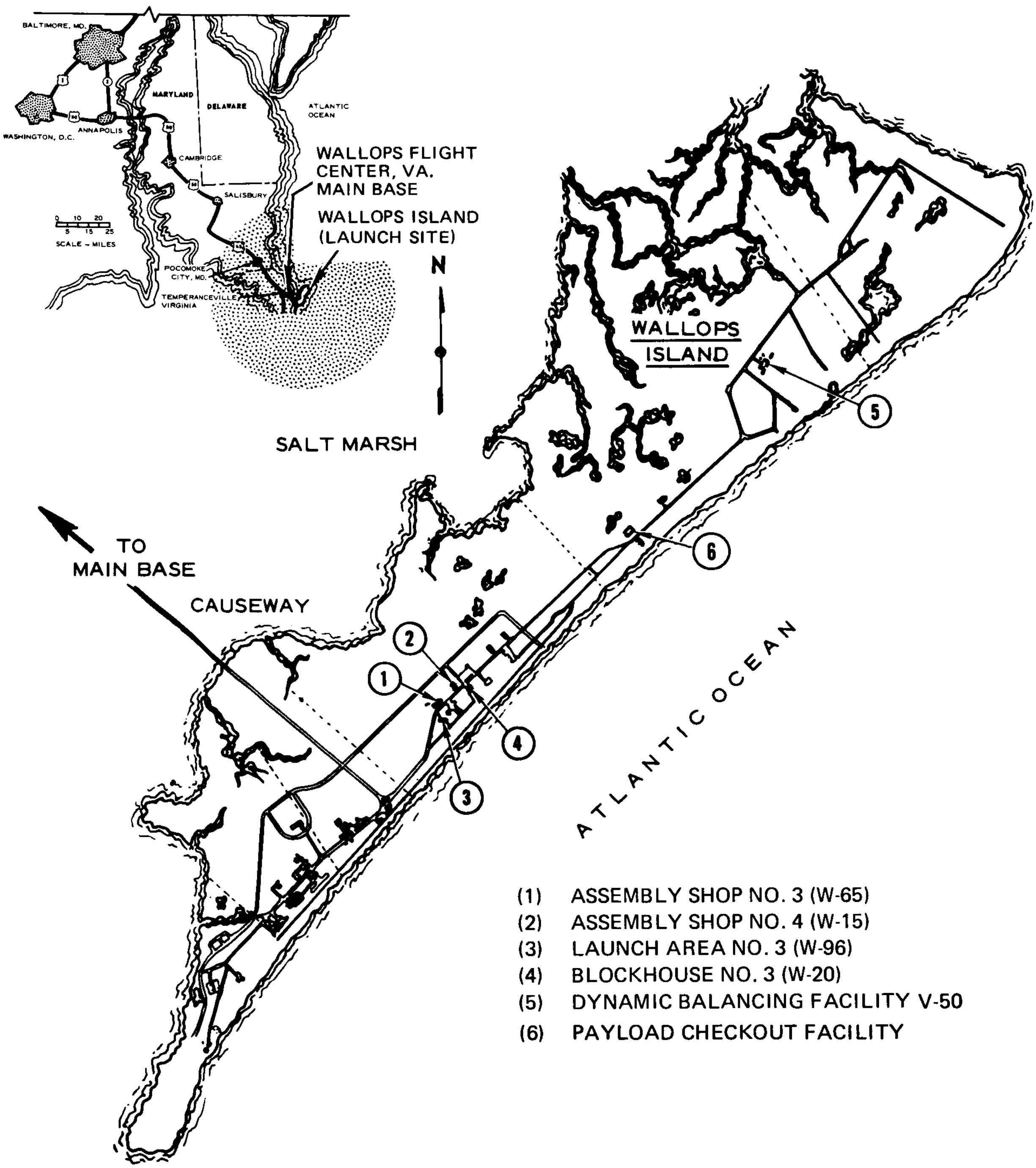
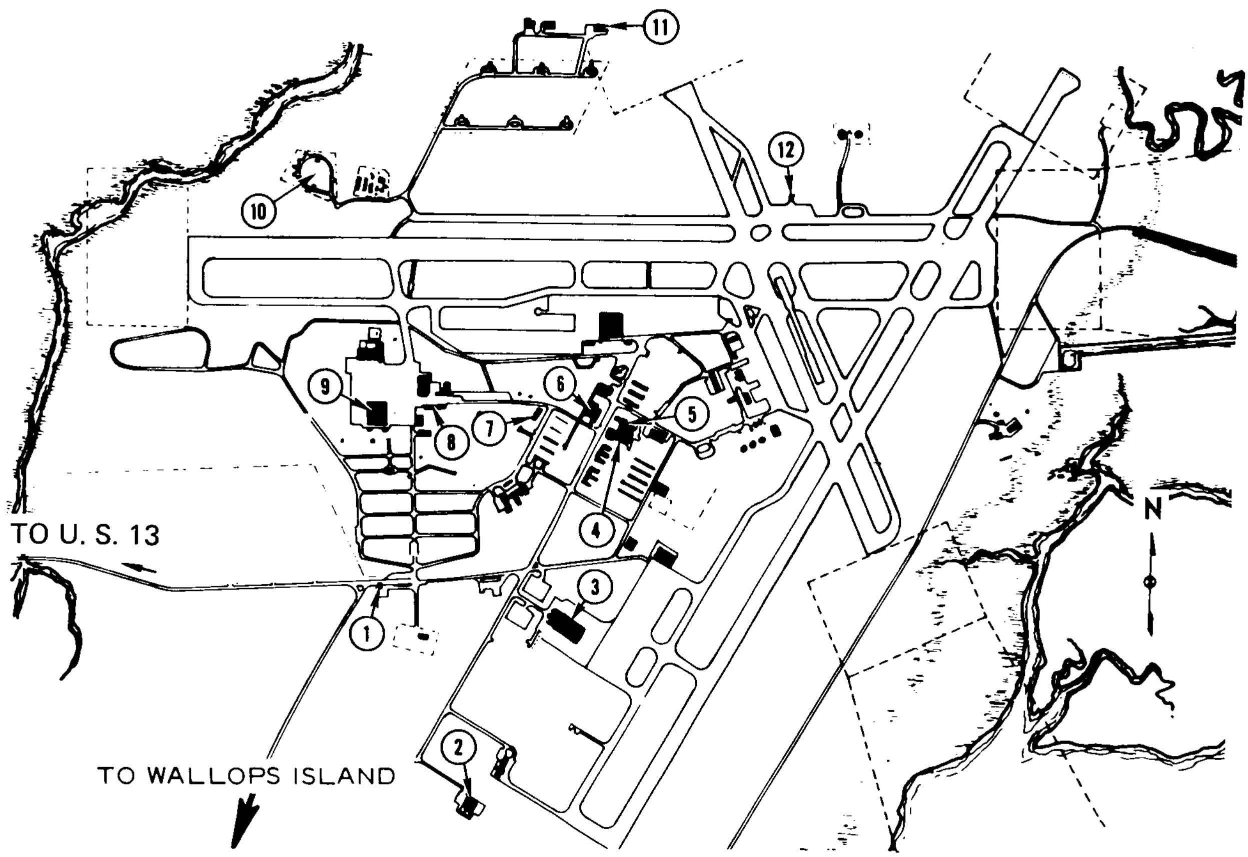


FIGURE IV–2 LOCALITY MAP OF WALLOPS FLIGHT CENTER, VIRGINIA

RANGE FACILITIES

Wallops Flight Center is located in a rural area where commercial sources of material and/or services are limited. In general, unless the availability of services and materials has been definitely established, the Scout user should assume that he must be completely self-sufficient. Figure IV-3 shows the general layout of the Wallops Flight Center main base.



(1) MAIN GATE AND GUARDHOUSE
(2) COMMUNICATIONS AND RECEIVING
(3) WALLOPS STATION RANGE CENTER
(4) CAFETERIA
(5) PHOTOGRAPHIC LABORATORY
(6) POST OFFICE
(7) WALLOPS STATION HEADQUARTERS
(8) SHIPPING AND RECEIVING BUILDING
(9) TECHNICAL SERVICES AREA
(10) EXPLOSIVE HANDLING AND STORAGE AREA "C"
(11) OPEN ROCKET STORAGE SHELTER
(12) EXPLOSIVE HANDLING AND STORAGE AREA "A"

FIGURE IV–3 WALLOPS FLIGHT CENTER, VIRGINIA

1.1.1 Assembly Building No. 3

The Scout Assembly Building No. 3 at Wallops Flight Center is located on the island near launch area No. 3 site. (See Figure IV-4 and IV-5).

The assembly building is a large L-shaped building used for receiving, storage, assembly, and checkout of the Scout vehicle. The building is divided into six bays, separated by heavy partitions of reinforced concrete. Bays 2 through 6 are used for individual step assembly buildup areas. After the individual steps have been assembled they are moved to bay 1 for vehicle buildup. A transporter trailer is the work platform on which the Scout assembly and checkout is performed. After the Scout checkout is completed the Scout vehicle is transported to the launcher.

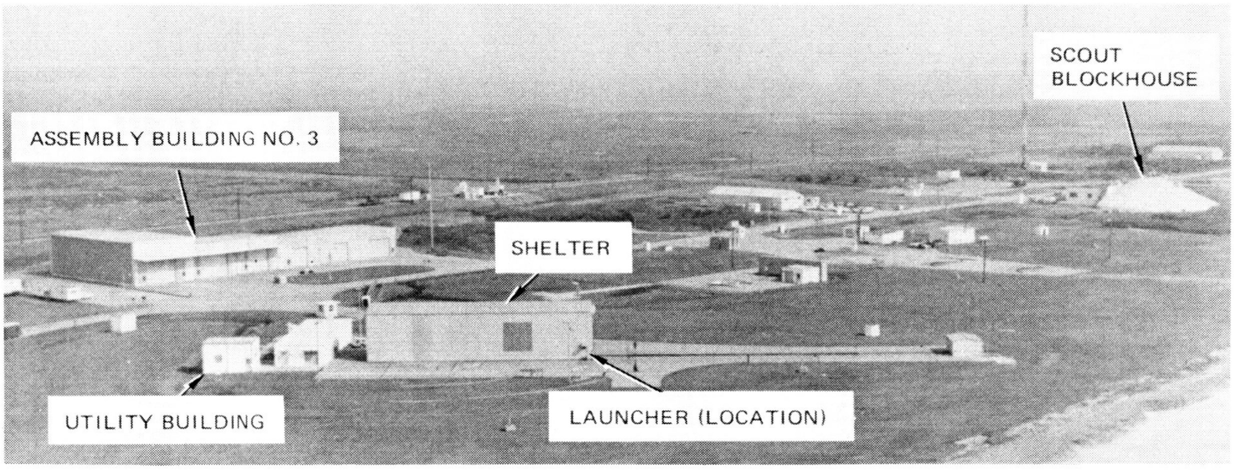


FIGURE IV–4 LAUNCH AREA NO. 3 ON WALLOPS ISLAND – WFC

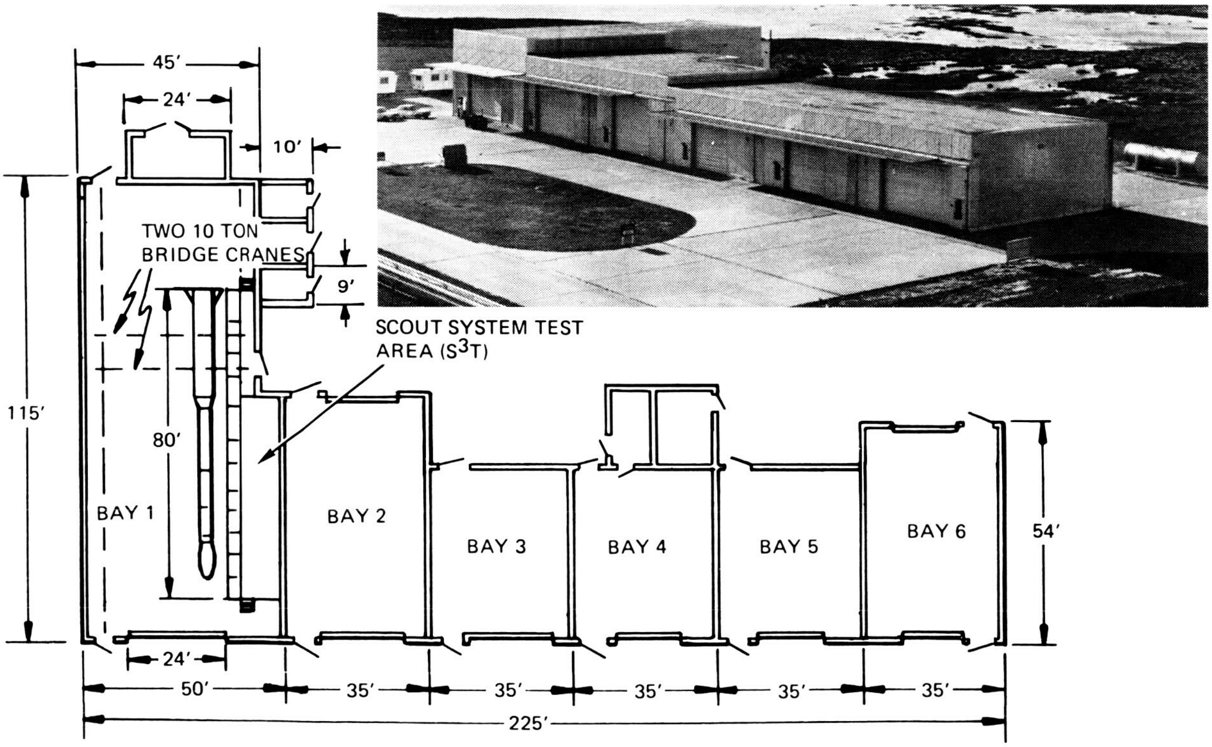


FIGURE IV–5 ASSEMBLY BUILDING NO. 3 ON WALLOPS ISLAND – WFC

1.1.2 Dynamic Balancing Facility

The Dynamic Balancing Facility (see Figure IV-6), located on the north end of Wallops Island, provides a means for dynamically balancing the payload, its mounting hardware, adapters, and fourth stage rocket motor.

The facility consists of three buildings, a blockhouse and two test buildings approximately 400 feet apart. Figure IV-7 illustrates the floor plans and the arrangement of the equipment. The Blockhouse contains a control room with all the necessary control, recording, and monitoring equipment to perform remote spin testing and computing of all unbalances in the buildup as well as the completely assembled fourth stage with the payload.

Communication between the control room and test buildings consists of three closed circuit TV systems and an intercom system. The TV systems, operable during balancing operations, provide the control room personnel with a complete view of the entire test building interior as well as a close-up of the balancing machines.

Test Building No. 1 contains a Gisholt (vertical) Balancing machine, and Test Building No. 2 contains a Trebel (vertical) Balancing machine. Both buildings contain the necessary equipment for monitoring, assembling, checkout, handling and weight of the disassembled and assembled fourth stage. Two hydraulic platforms are provided with each balancing machine, which permit 360 degree access to the payload.

The Gisholt balancing machine (see Figure IV-8), is capable of balancing a fourth stage weighing up to 2,000 pounds, with a maximum diameter of 43 inches at speeds of 150 – 1000 rpm.

The Trebel balancing machine (see Figure IV-9) is a hard bearing, permanently calibrated, machine capable of balancing a fourth stage weighing up to 6,000 pounds, with a maximum diameter of 10 feet at speeds from 30 to 1000 rpm.

It should be noted that the payload must be mounted vertically on these machines, and that the Payload Agency is responsible for providing handling rigs. These rigs will also be useful in general handling of the payload when it must be moved from one area to another.

1.1.3 Clean Room Facility

A 20 X 20 X 8 foot class 10,000 clean room, located in Bay 6 of assembly shop 3 (W-65), is available for spacecraft operations requiring a clean environment. This facility is capable of delivering vertical downward flow class 100 air and maintaining the required temperature ± 2^{o}F and the relative humidity below 45 percent. Personnel entry is through an air lock passage, while equipment entry/exit can be provided through an 8 X 8 foot doorway.

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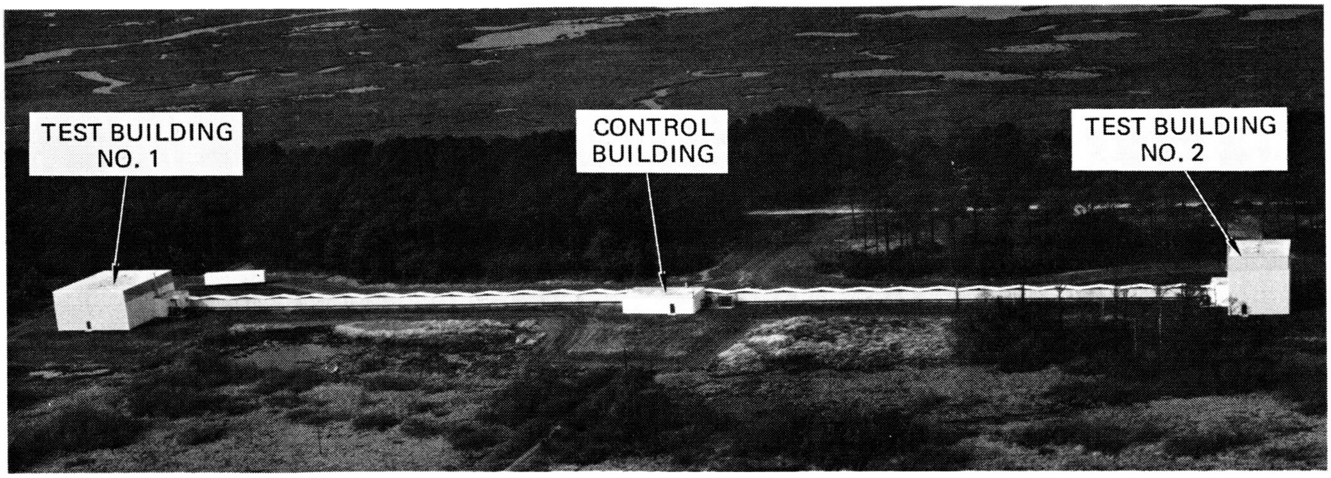
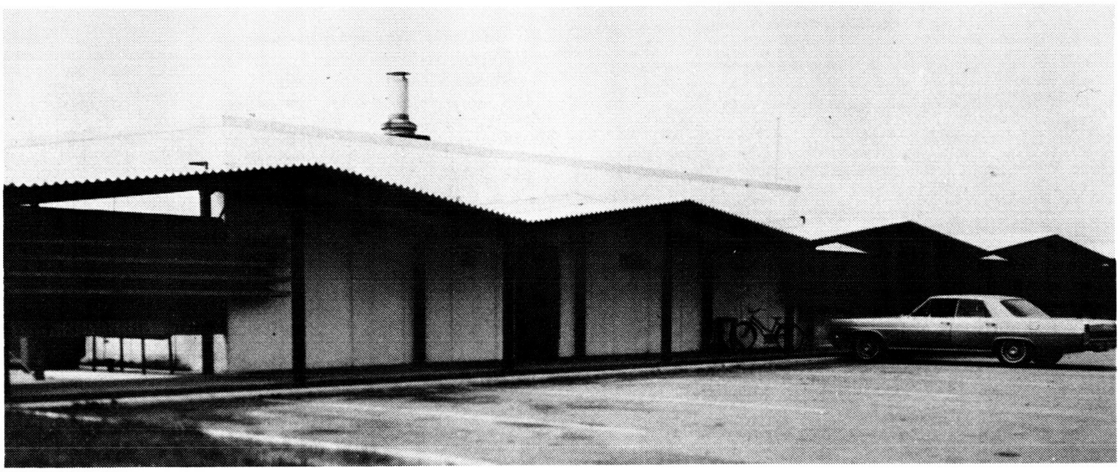
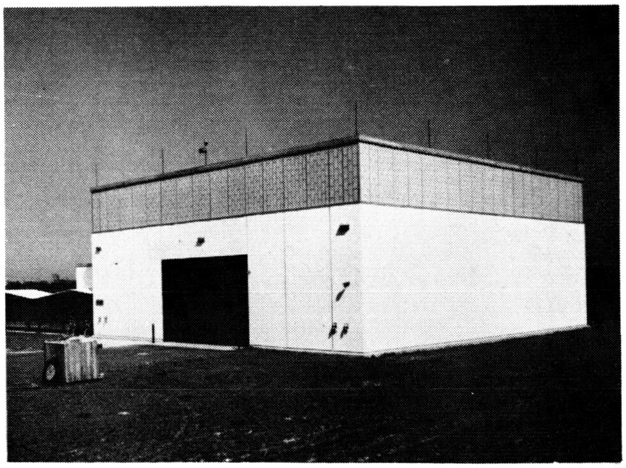


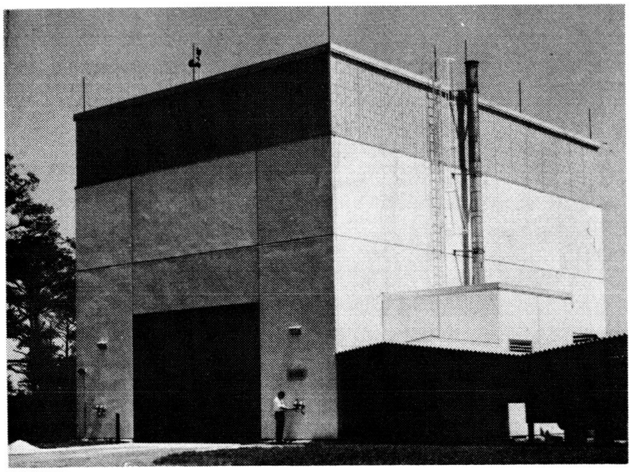
AERIAL VIEW LOOKING NORTHWEST



CONTROL BUILDING (V–50)

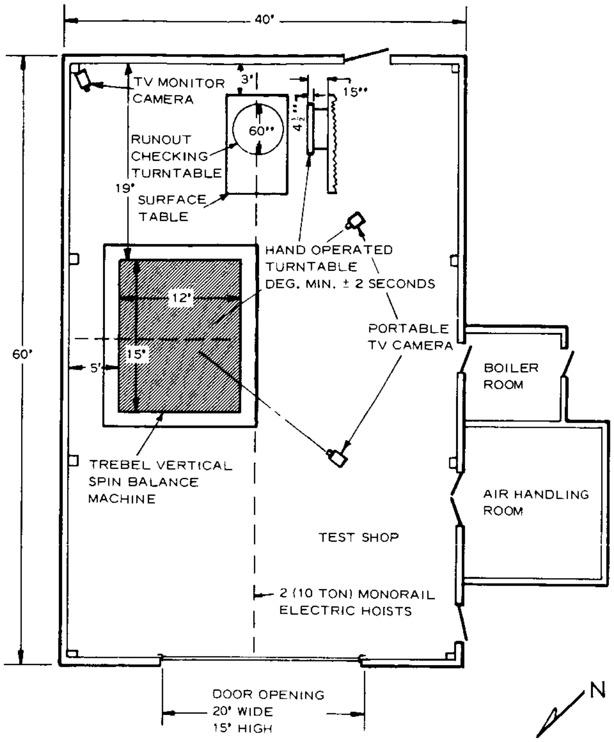


TEST BUILDING NO.1 (V–45)

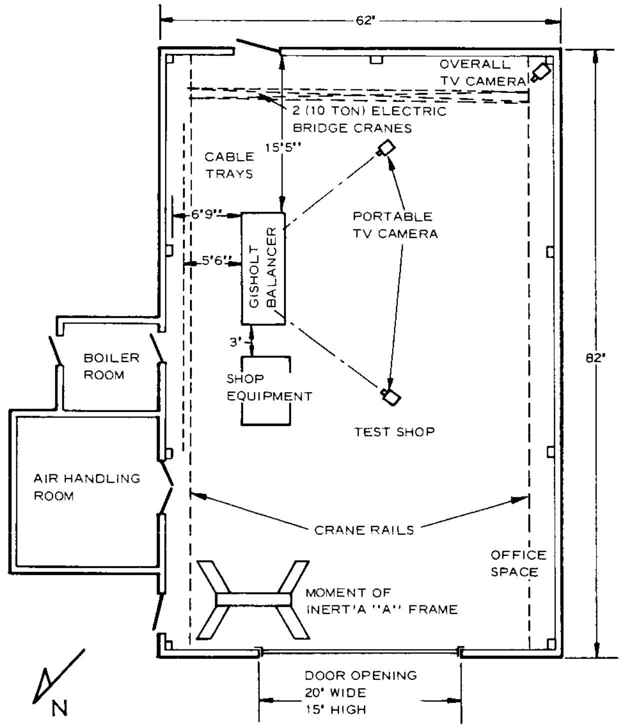


TEST BUILDING NO.2 (V–55)

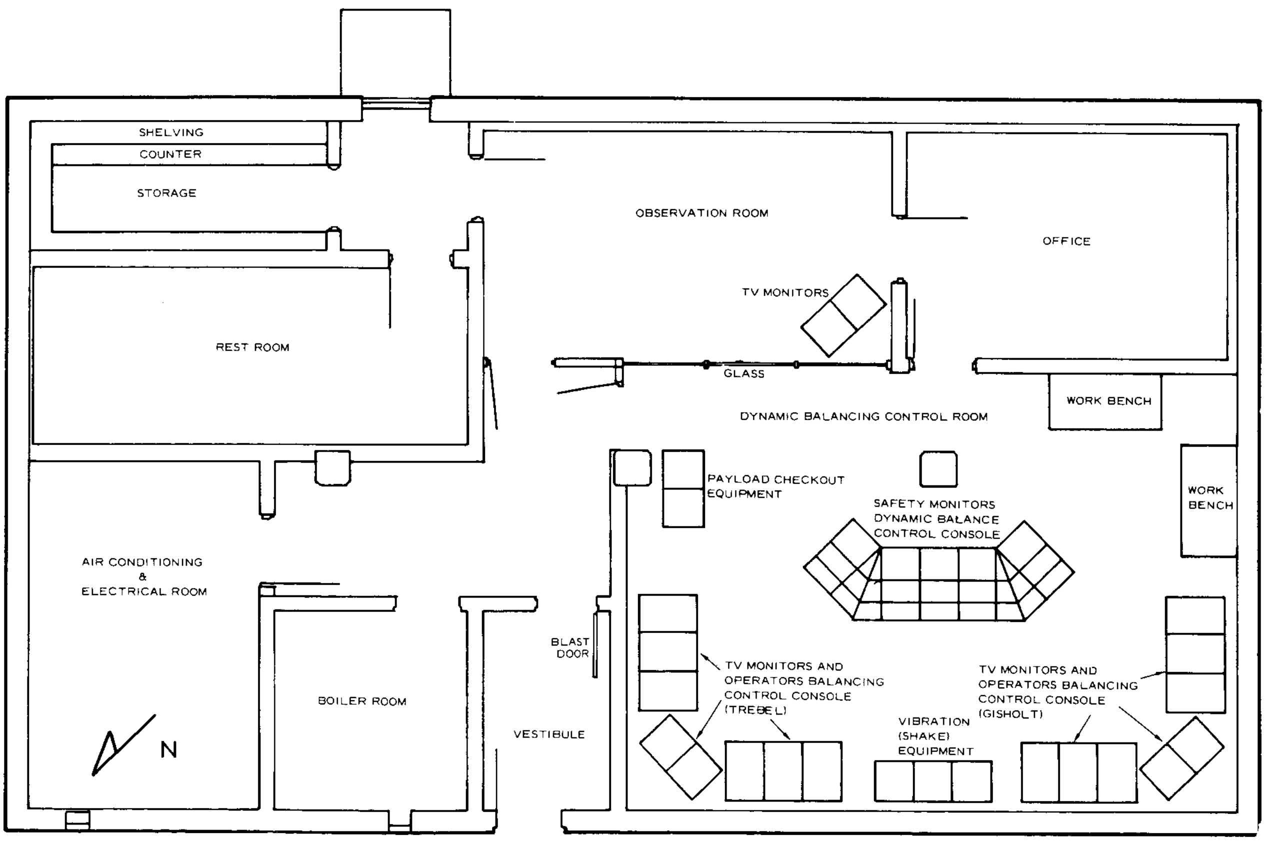
FIGURE IV–6 DYNAMIC BALANCING FACILITY ON WALLOPS ISLAND – WFC



TEST BUILDING NO. 2 (V–55)



TEST BUILDING NO. 1 (V–45)



DYNAMIC BALANCE CONTROL BUILDING – V–50 (BLOCKHOUSE)

FIGURE IV–7 DYNAMIC BALANCE FACILITY FLOOR PLANS – WFC

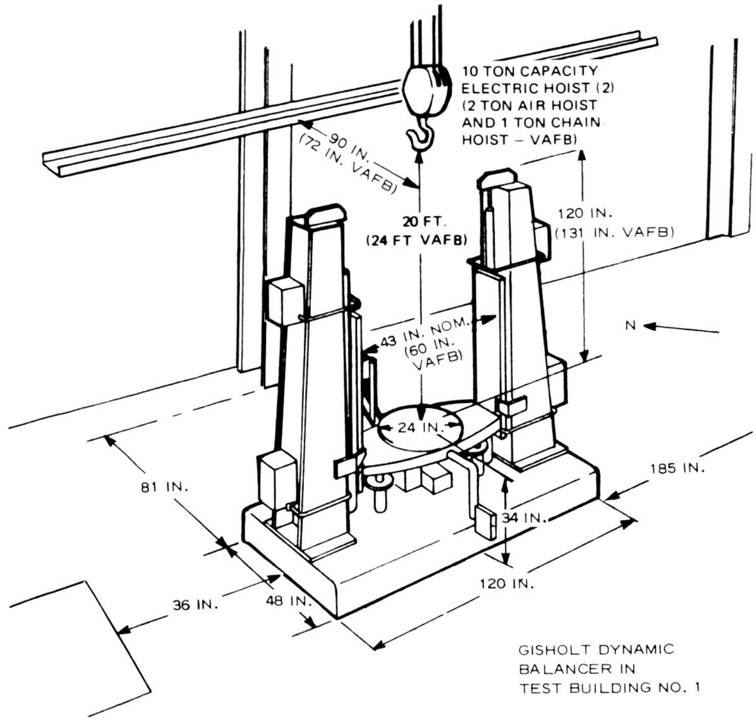


FIGURE IV–8 TEST BUILDING NO. 1 – BALANCING EQUIPMENT – WFC

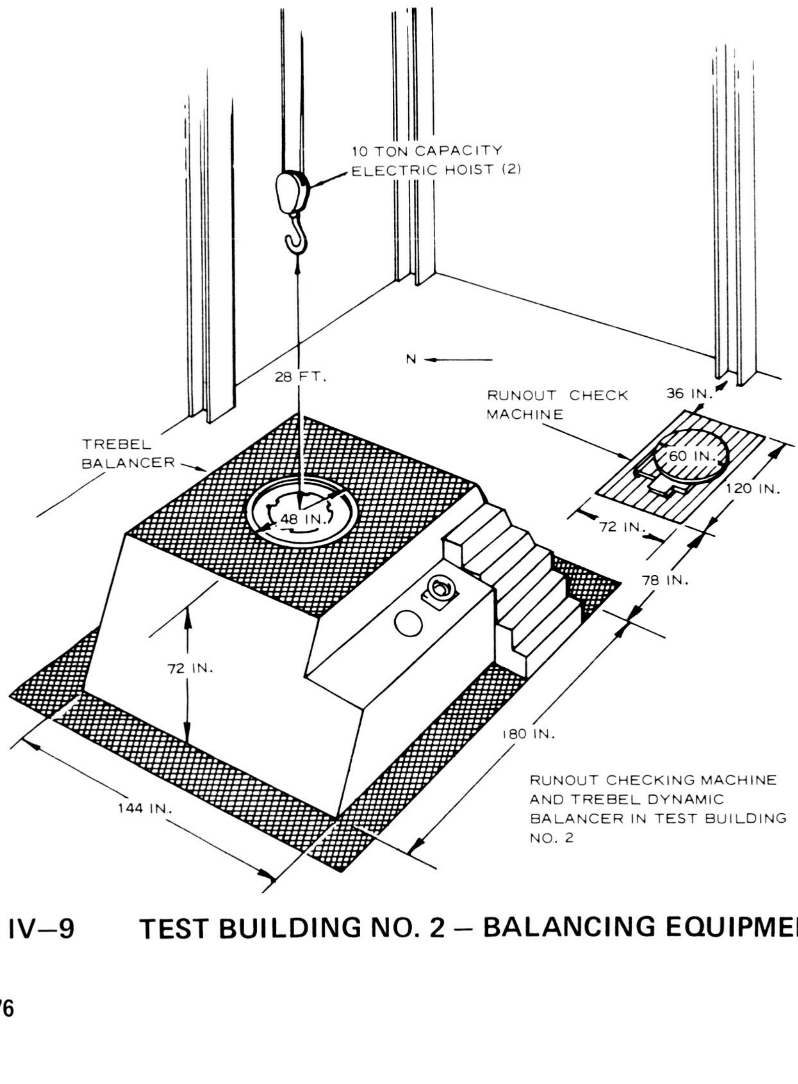


FIGURE IV–9 TEST BUILDING NO. 2 – BALANCING EQUIPMENT – WFC

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1.1.4 Payload Work Facility

The payload assembly and checkout facilities, (see Figure IV-10) provided for Payload Agency use, are temperature controlled masonry buildings located midway between the Scout launch area No. 3 and the Spin Test Facility. The Inert Bay and Office Building (V-25) provides approximately 1400 square feet of work space. The Hot Bay Building (V-26) provides approximately 500 square feet of work space. The buildings are equipped with large doors, numerous 110 VAC and 220 VAC electrical outlets, workbenches, office equipment and chain fall hoists.

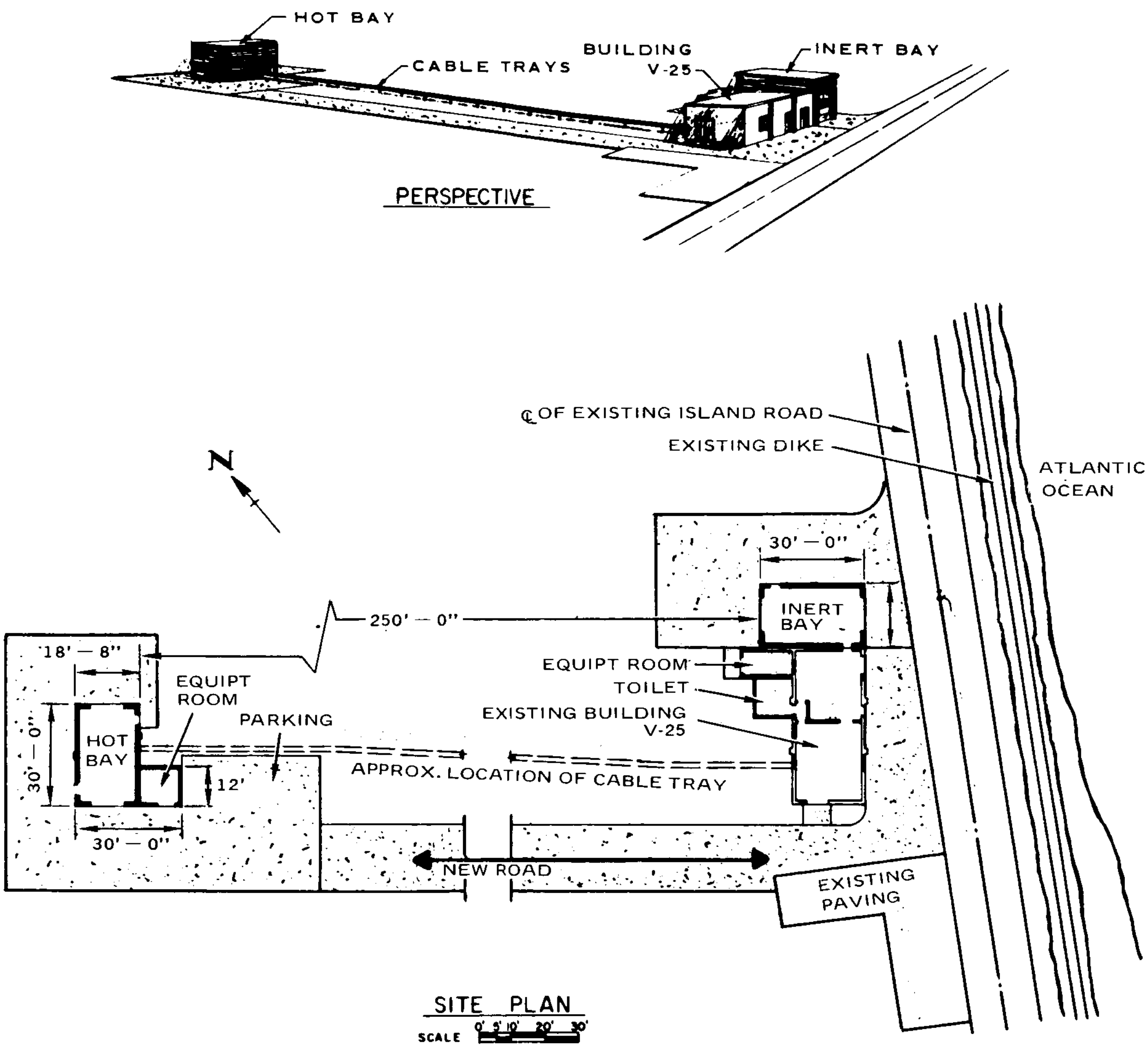

FIGURE IV–10 PAYLOAD WORK FACILITY – WFC

1.2 VANDENBERG AIR FORCE BASE (VAFB)

The launch site for Scout vehicles on the West coast of the United States is located at Vandenberg AFB, California. The Mission Working Group meeting for Vandenberg AFB launches are usually at the base, at which time a tour of facilities is provided to the Scout user. The Scout support facilities at South Vandenberg consist of the NASA Spacecraft Laboratory, Ordnance Assembly Building (OAB), and Spin Test Facility (STF). A locality map and site plan for South Vandenberg are shown in Figure IV-11.

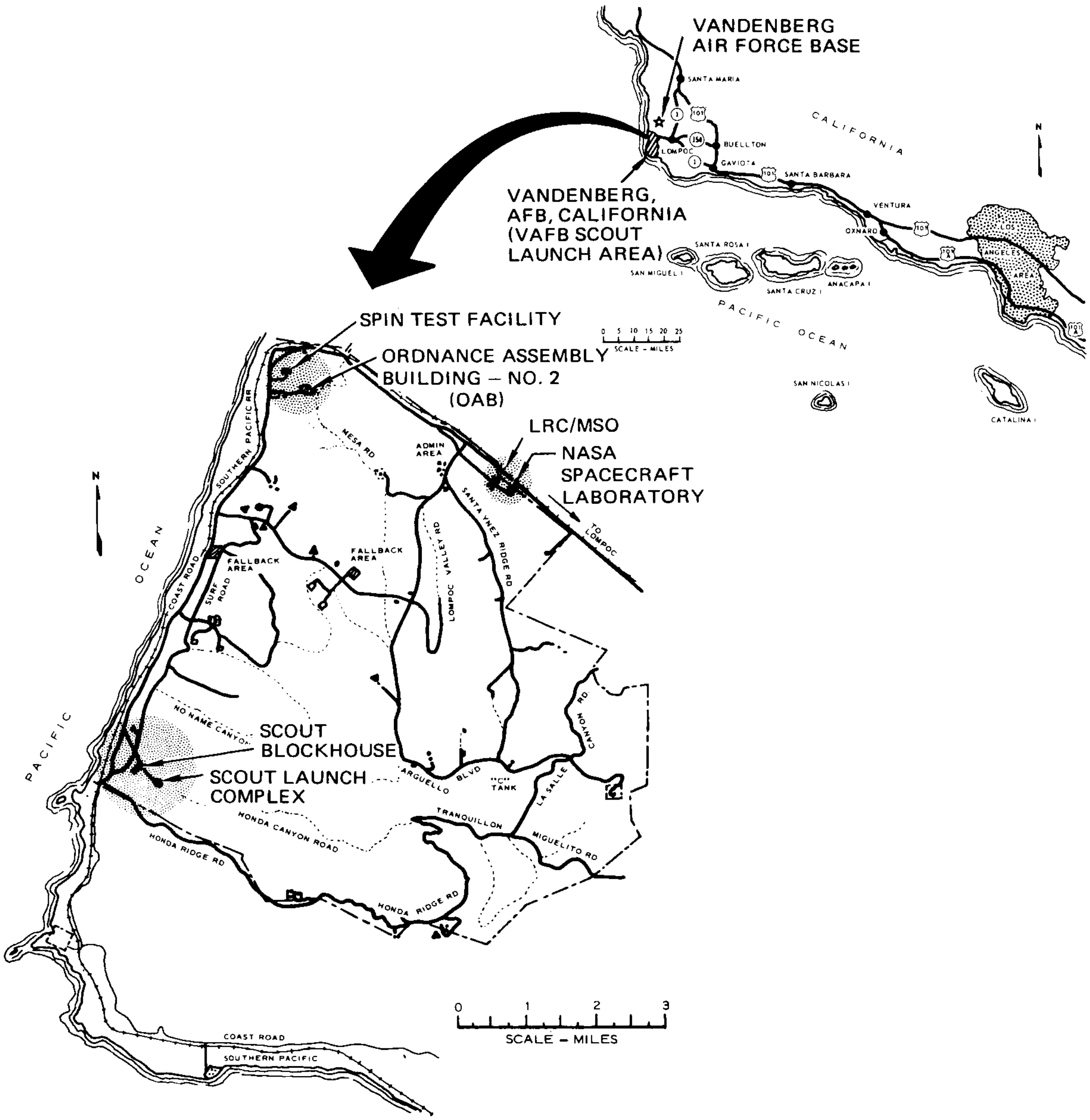

FIGURE IV–11 LOCALITY MAP AND SITE PLAN OF SOUTH VANDENBERG AFB, CALIFORNIA

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1.2.1 NASA SPACECRAFT LABORATORY BUILDING

The NASA Spacecraft Laboratory Building, Bldg. 836 (see Figure IV-12), is located in the Administration area of South Vandenberg AFB. This buidling houses areas available for spacecraft agency offices in the west half of the Building, and the NASA/GSFC Spacecraft Laboratory in the east half. The Spacecraft Laboratory maintains a clean room which can be made available by coordinating with the Mission Working Group. The clean room is temperature controlled at 70 ± 5°F with a relative humidity at approximately 50%. Particle control is comparable to that in an average light electronic manufacturing area.

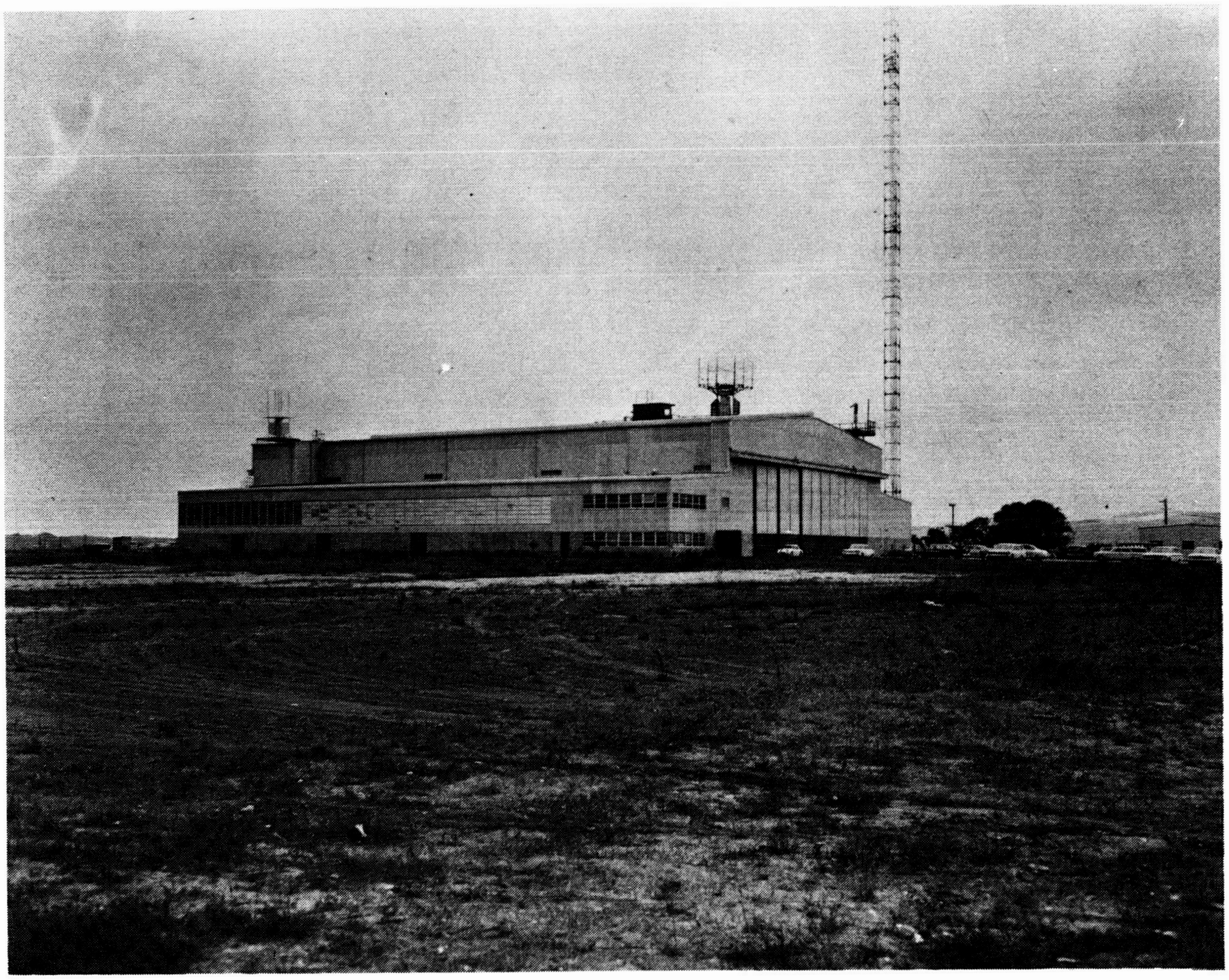

FIGURE IV–12 NASA SPACECRAFT LABORATORY BUILDING – VAFB

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1.2.2
Ordnance Assembly Building — OAB

The Ordnance Assembly Building at South Vandenberg AFB is used for the Scout vehicle. It is called the OAB No. 2 (Building 960) (see Figure IV-13). This building is used for motor checkout, Scout vehicle assembly, buildup, and systems checkout.

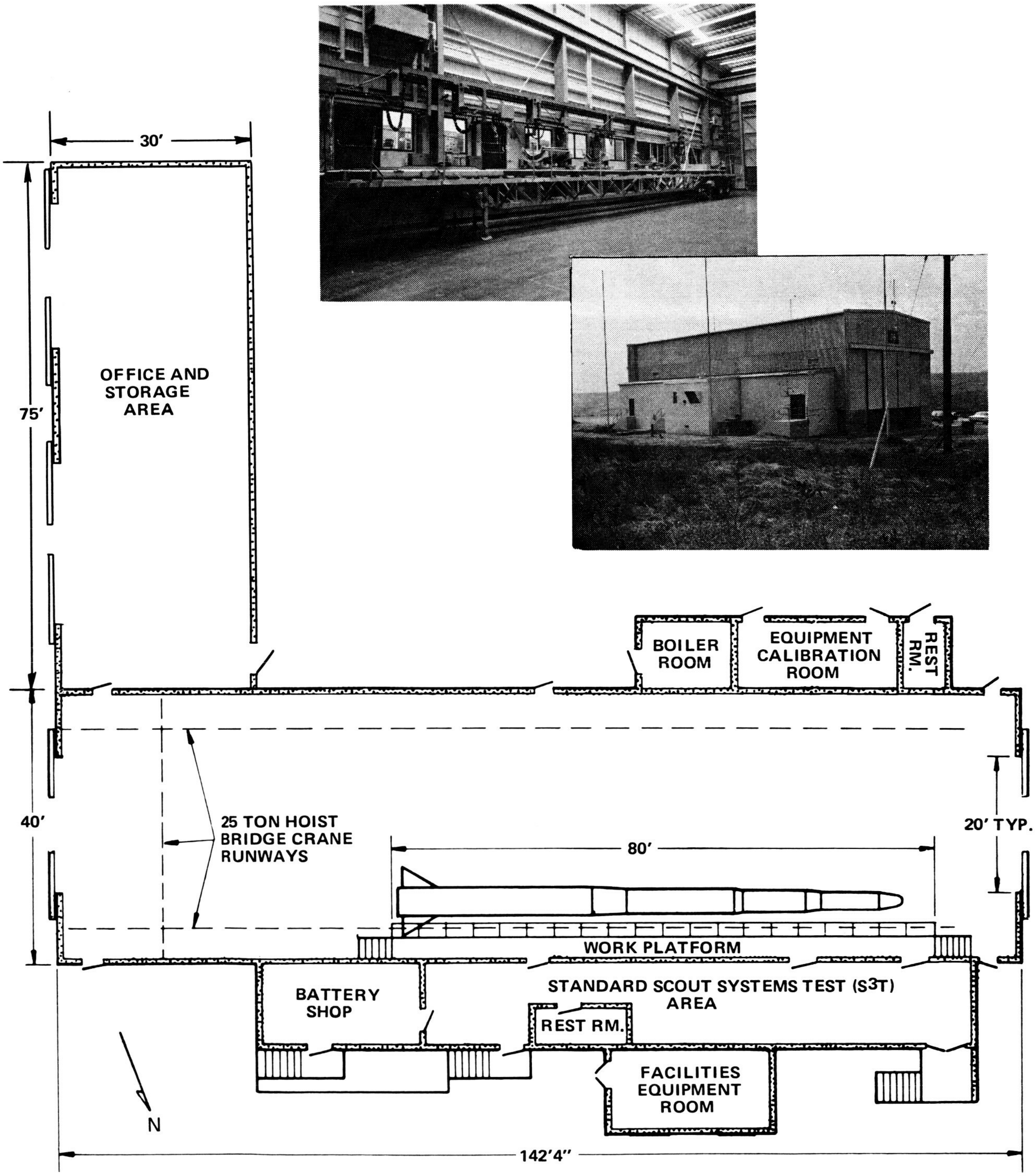


FIGURE IV–13 ORDNANCE ASSEMBLY BUILDING – VAFB

1.2.3 Spin Test Facility

The Spin Test Facility at South Vandenberg AFB consists of a control building, spin balance building, and spin test operations support building (see Figure IV-14 and IV-17).

a. Control Building No. 995

The Control Building is heavily reinforced and contains the necessary equipment to perform remote spin testing and monitoring by closed circuit TV. The control unit and floor plan are shown in Figure IV-16.

b. Spin Balance Building No. 996

The Spin Balance Building houses the Gisholt balancing machine (see Figure IV-8 for physical dimensions) and necessary equipment for assembly, checkout handling, and weighing of the payload, and equipment to compute all unbalances in the buildup of the fourth and fifth stage as well as the completely assembled fourth and fifth stage. It is also heavily reinforced to protect the area around the facility. See Figure IV-15 for floor plan and interior view of this building. Handling rigs for the payload are the responsibility of the Payload Agency.

c. Spin Test Operations Support Building No. 997

The Spin Test Operations Support Building (see Figure IV-17), located adjacent to the Control and Spin Balancing Buildings, consists of office space and work area. The work area can be made available for payload use by coordinating with the VAFB Scout Launch Team. The building is not an approved area for ordnance items.

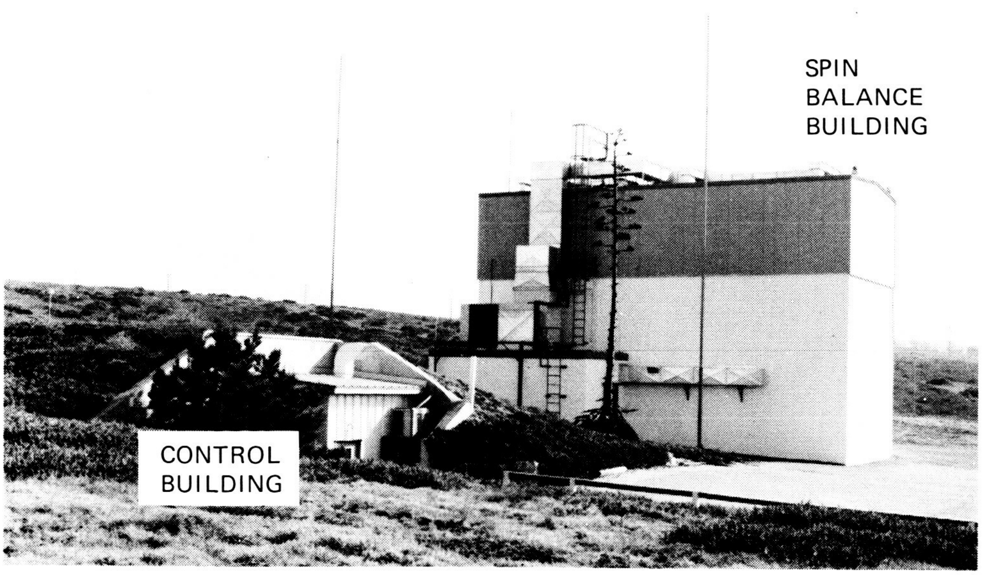
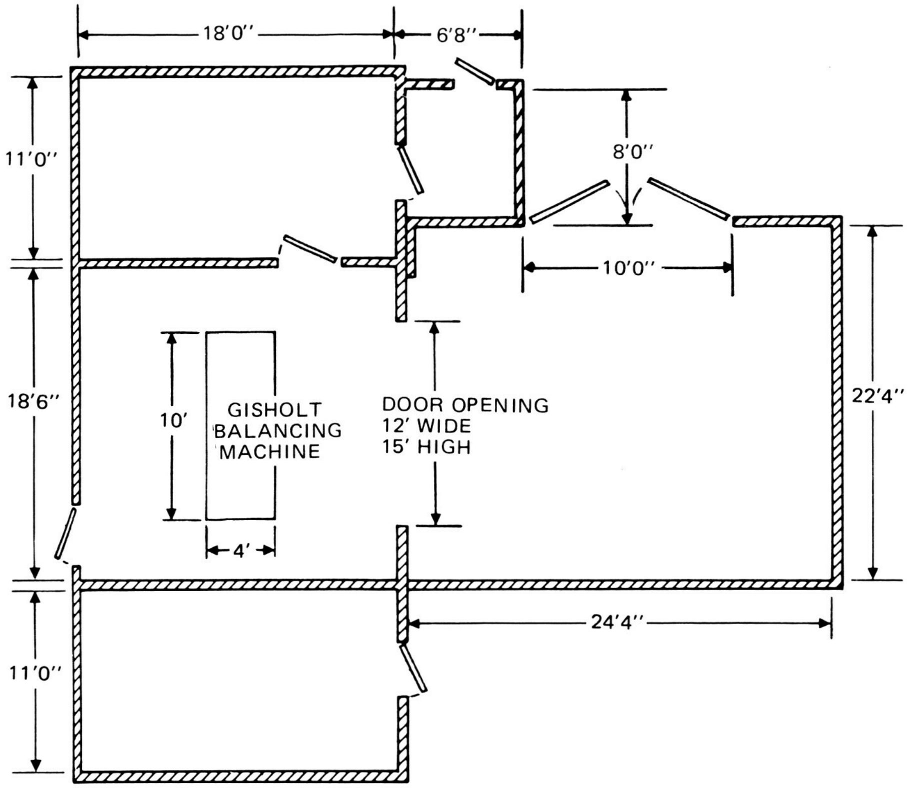
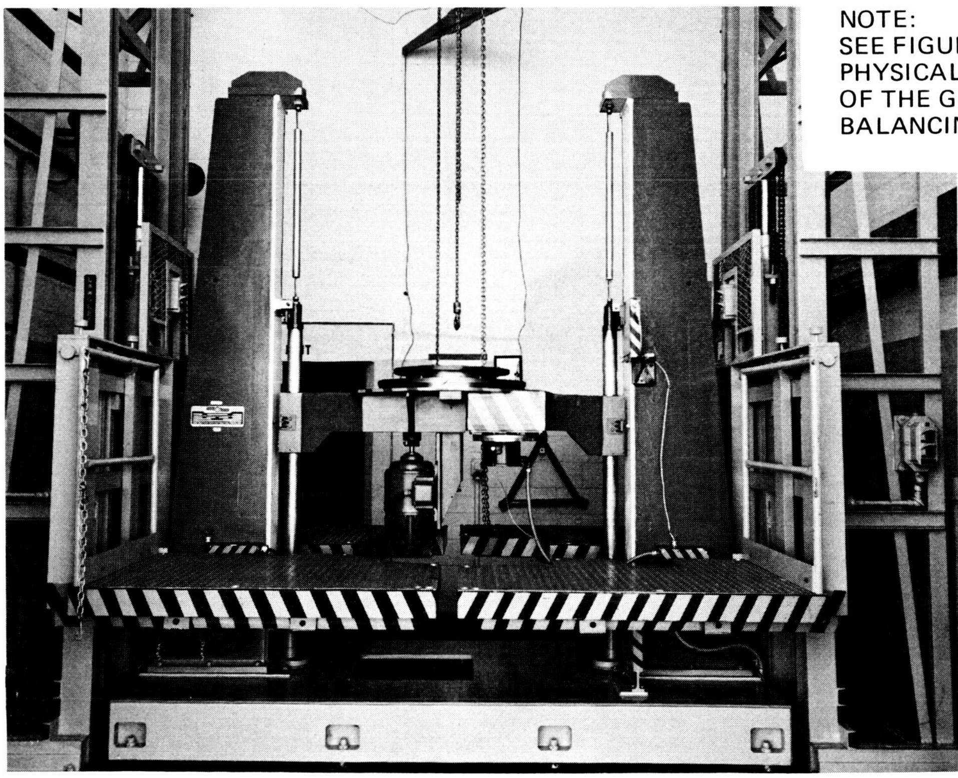


FIGURE IV–14 SPIN TEST FACILITY – VAFB





NOTE:
SEE FIGURE IV–8 FOR PHYSICAL DIMENSIONS OF THE GISHOLT BALANCING MACHINE

FIGURE IV–15 SPIN BALANCING BUILDING FLOOR PLAN AND GISHOLT BALANCING MACHINE – VAFB

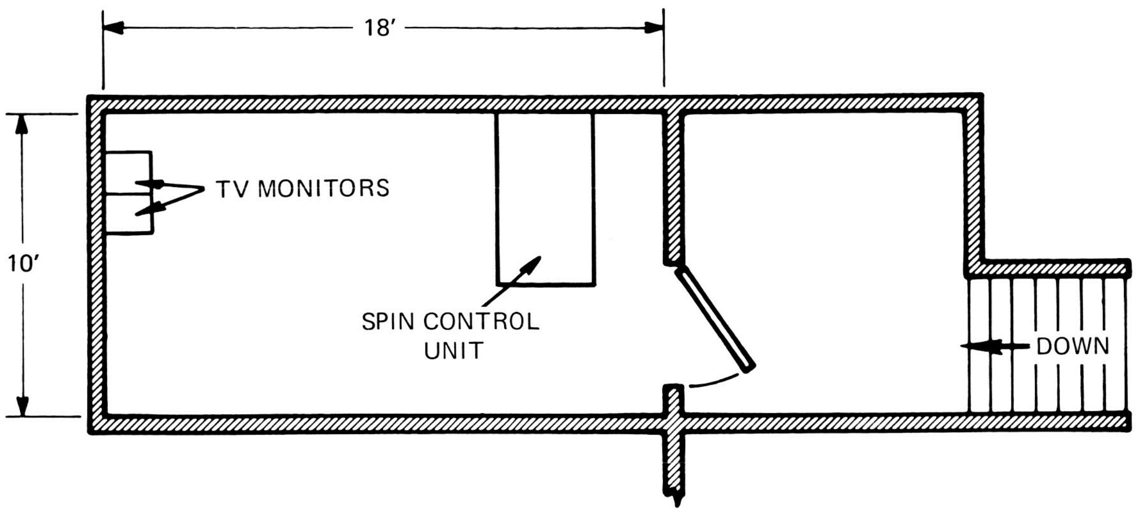
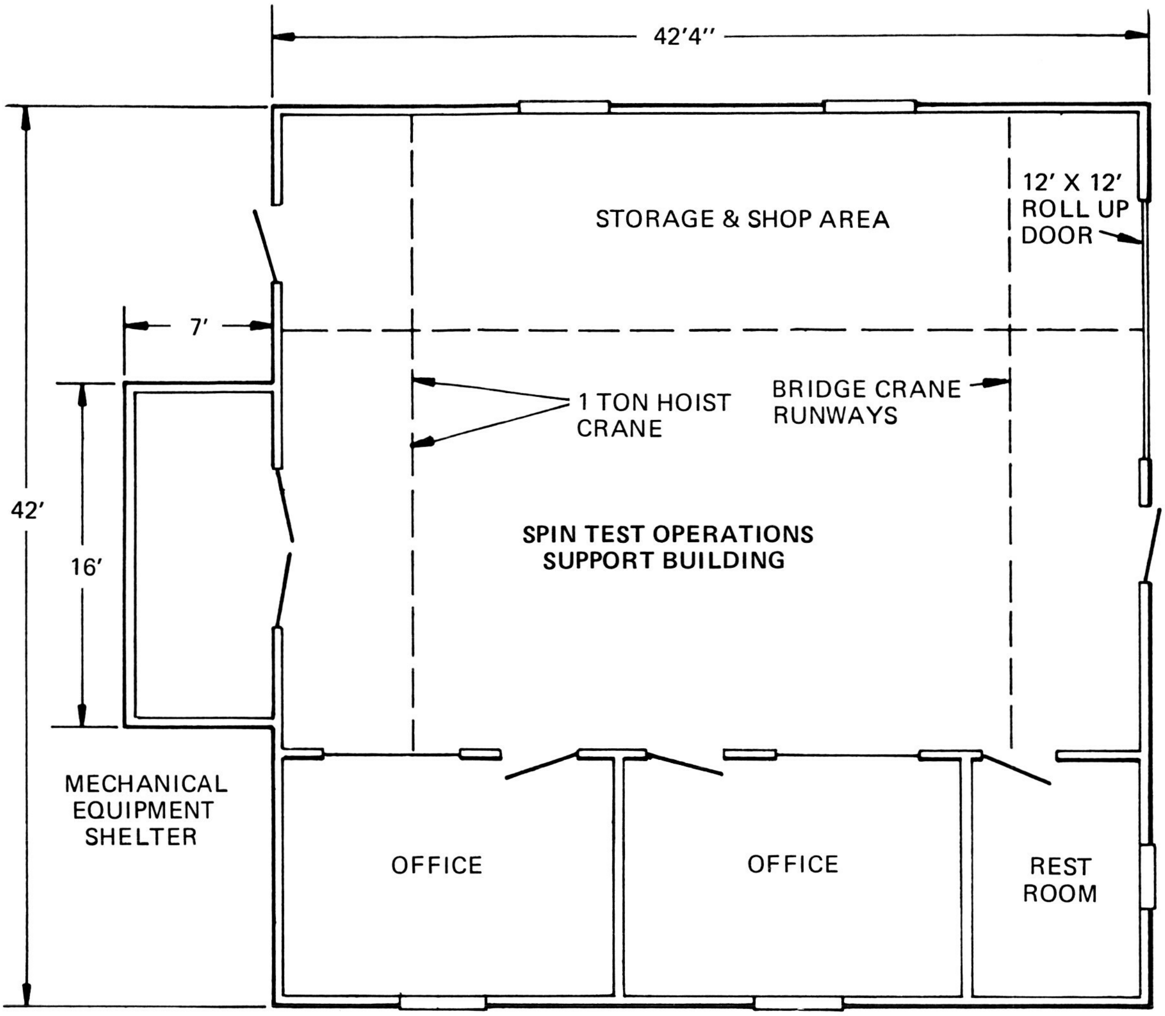


FIGURE IV–16 SPIN TEST CONTROL BUILDING AND CONTROL UNIT – VAFB



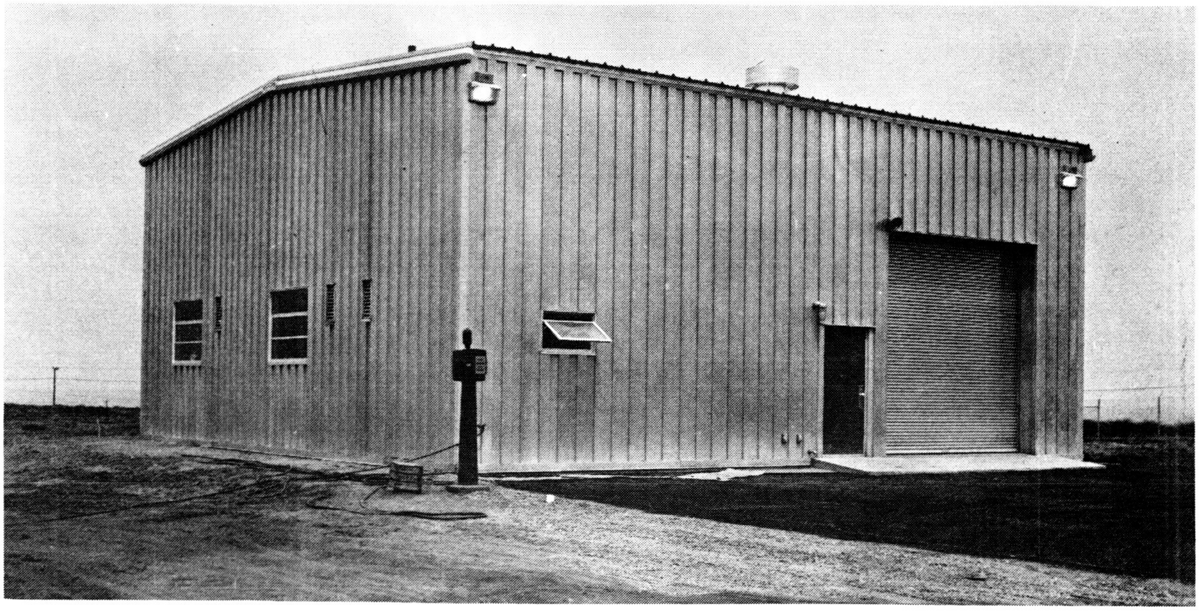


FIGURE IV–17 SPIN TEST OPERATIONS SUPPORT BUILDING – VAFB

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1.3 SAN MARCO EQUATORIAL MOBILE RANGE

The San Marco Equatorial Mobile Range is located in Ngwana Bay, formerly Formosa Bay, of the Indian Ocean off the east coast of Kenya, Africa. The launch complex consists of two floatable platforms. The San Marco platform serves as the launching platform. The Santa Rita platform serves as the focal point of range operations. Logistic support is provided from a base camp located on the mainland nearby. A locality map of the range is shown in IV-18.

The range is accessible by air or sea from all parts of the world. Nairobi is the port-of-entry for air passengers and cargo in Kenya. Mombasa is the port-of-entry for sea shipments. The Scout rocket motors and other hazardous material shipped by sea are normally delivered directly to the San Marco platform.

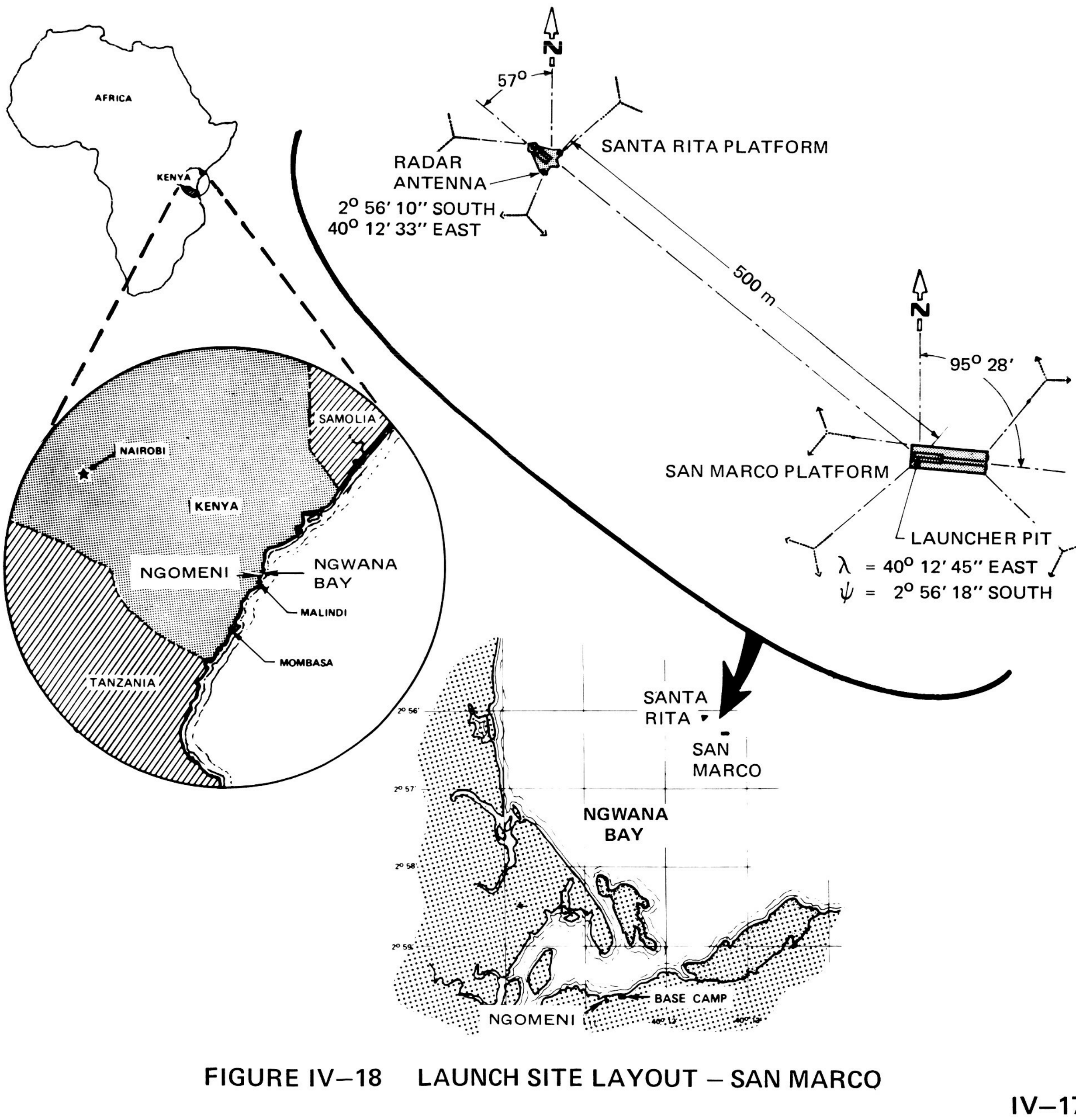

FIGURE IV–18 LAUNCH SITE LAYOUT – SAN MARCO

1.3.1 San Marco Platform

The San Marco platform (see Figure IV-19) serves as the launching platform and contains the launcher and ground support equipment necessary for vehicle assembly, checkout and launch. Vehicle assembly and checkout is performed inside the launcher shelter on the transporter.

Spacecraft assembly and pre-installation checks can be performed in Shelter S-3, an environmental controlled room. The 17 foot by 25 foot shelter is located on the deck of the San Marco platform. Should the spacecraft require it, the room is capable of meeting the requirements of a clean room.

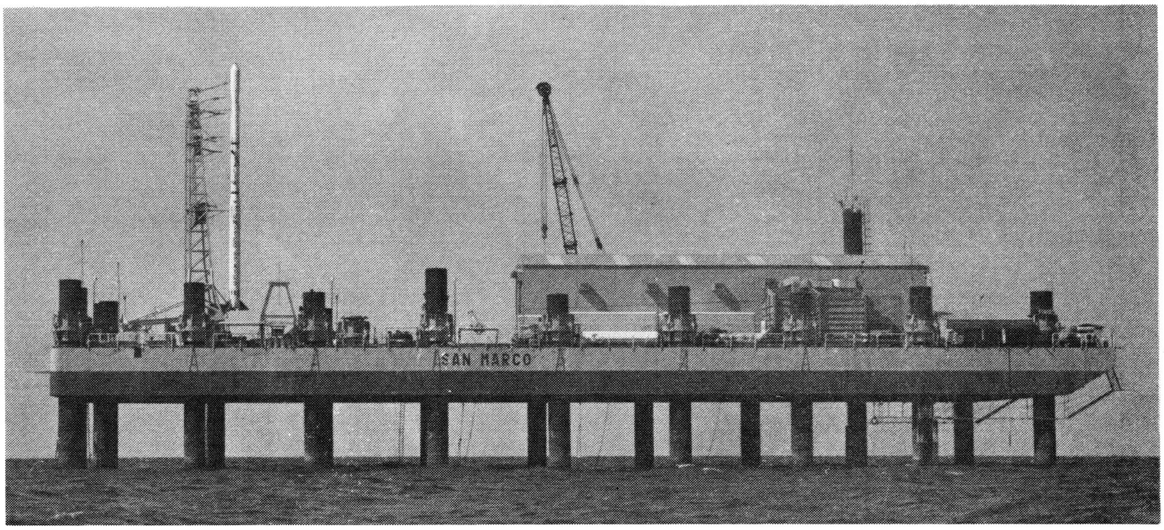

FIGURE IV–19 SAN MARCO PLATFORM

1.3.2 Santa Rita Platform

The Santa Rita platform (see Figure IV-20) serves a combined range control, blockhouse and logistics facility. The Range Control Center is located in a compartment below deck. Vehicle launch control is conducted from a trailer-van blockhouse located on the main deck.

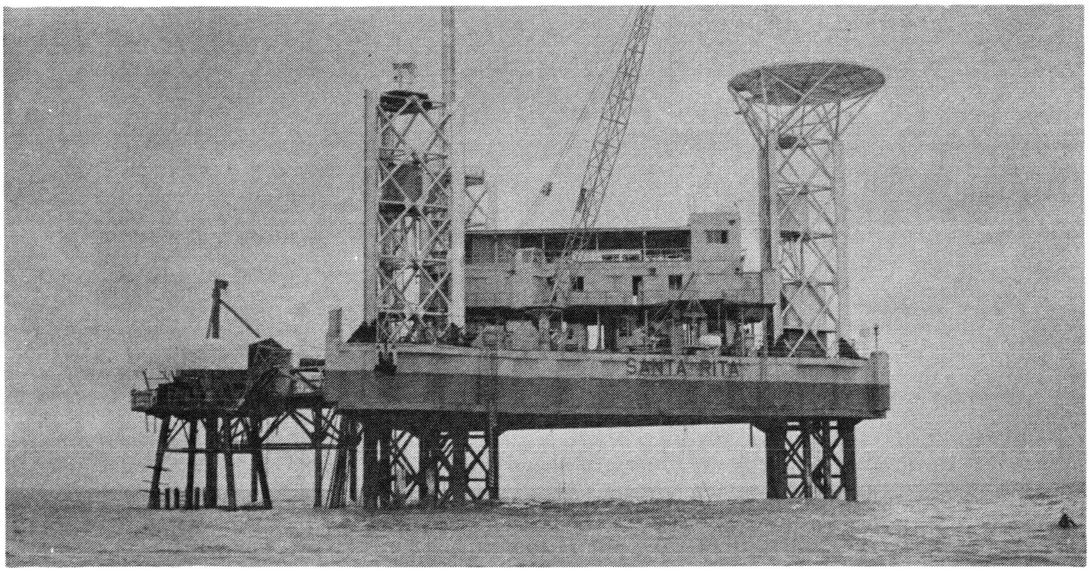

FIGURE IV–20 SANTA RITA PLATFORM

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SCOUT STANDARD LAUNCH COMPLEX

2

2.0 SCOUT STANDARD LAUNCH COMPLEX

The major components of the Scout Standard Launch Complex are the launcher, the movable shelter, and the blockhouse. The configuration of the launch complex at each range is functionally identical but differs physically. Figure IV-21 shows the launcher in a horizontal position with shelter rolled back and with vehicle on transporter.

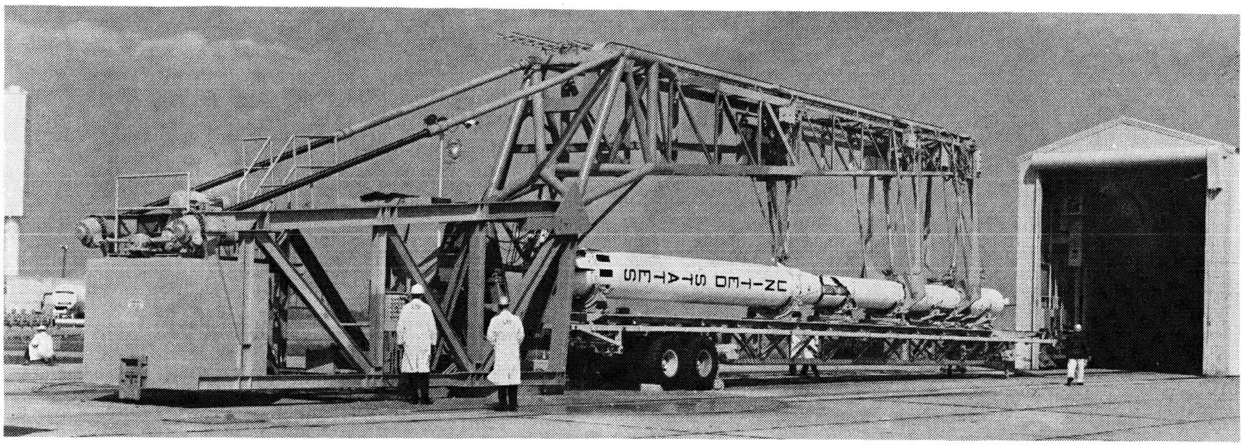

FIGURE IV–21 SCOUT LAUNCHER, TRANSPORTER AND SHELTER

2.1 SHELTER DESCRIPTION

The movable shelter (Figure IV-22) provides controlled environmental protection for pre-launch operation of the Scout vehicle. It is a galvanized steel building, mounted on rails, with power operated doors on each end and work platforms along each side. The Shelter is approximately 140 feet long, 24 feet wide, and 31 feet high.

Interior lights to illuminate the complete vehicle and work spaces are provided. Service outlets are provided along each wall as well as an air distribution system for power tools. Payload requirements for use of Shelter services are coordinated through the Mission Working Group.

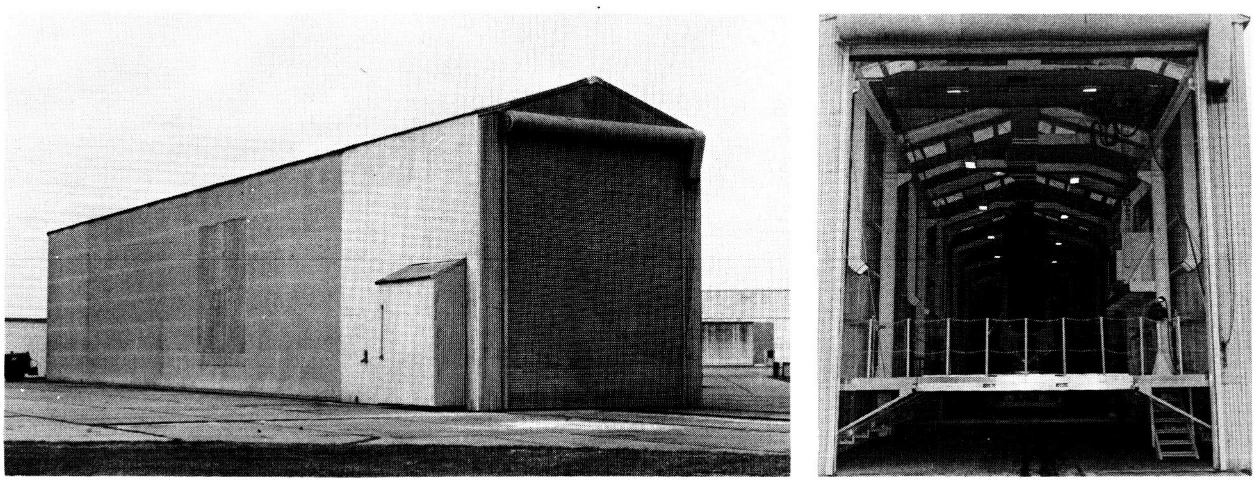

FIGURE IV–22 SCOUT MOVABLE SHELTER

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ALL PAYLOAD CHECKOUT EQUIPMENT, USED IN THE SHELTER, MUST HAVE WRITTEN APPROVAL BY RANGE SAFETY PRIOR TO ARRIVAL AT THE LAUNCH SITE.

2.1.1 Environment – Shelter

Ducts and louver-type outlets are provided for adequate distribution of thermostatically controlled air. For the purpose of motor conditioning, the shelter is maintained at 70°F up to three hours before launch. Humidity is not controlled beyond the capacity of a standard air conditioning system.

– Heatshield (WFC, VAFB, SM)

Further environmental control may be provided by placing the heatshield around the payload and purging it with air from a launcher-mounted Hilsch-tube environmental control system. This system can supply cooled, or heated, dried and filtered oil-free air with a maximum dew point of 35°F. The capabilities of this system, based on standard temperature (70°F) and pressure (14.7 psia) are defined by the shaded areas of Figure IV-23. These conditions apply to the air as supplied to the heatshield. Actual environmental conditions inside the heatshield will be dependent upon the heat dissipated, and the volume displaced, by the payload. The heatshield inlet air temperature can be controlled and monitored from the blockhouse.

– Heatshield (WFC, VAFB)

An optional environmental control system has been incorporated at WFC and VAFB for payloads with environmental requirements exceeding the capability of the Hilsch-tube system. This high volume system is located in a building adjacent to the launch pad and supplies conditioned air to the heatshield by means of insulated ducting routed up the launcher.

This system provides air with a dew point of 35°F at flow rates between 50 and 200 SCFM. The heatshield inlet air temperature can be controlled and monitored from the blockhouse within the range of 55 to 80°F (Figure IV-23A). The system is designed to filter out 99.97% of all particles larger than 0.3 micron. Testing during a spacecraft field operation, at 180 SCFM, counted 40 particles greater than 0.5 micron during a four minute interval. This test shows the system is capable of maintaining a class 100 level in accordance with MIL-STD-209 defintions.

Note

Since the heatshield encloses both the payload and fourth stage motor, and motor conditioning is required, temperature requirements other than 70°F must be approved by the Mission Working Group.

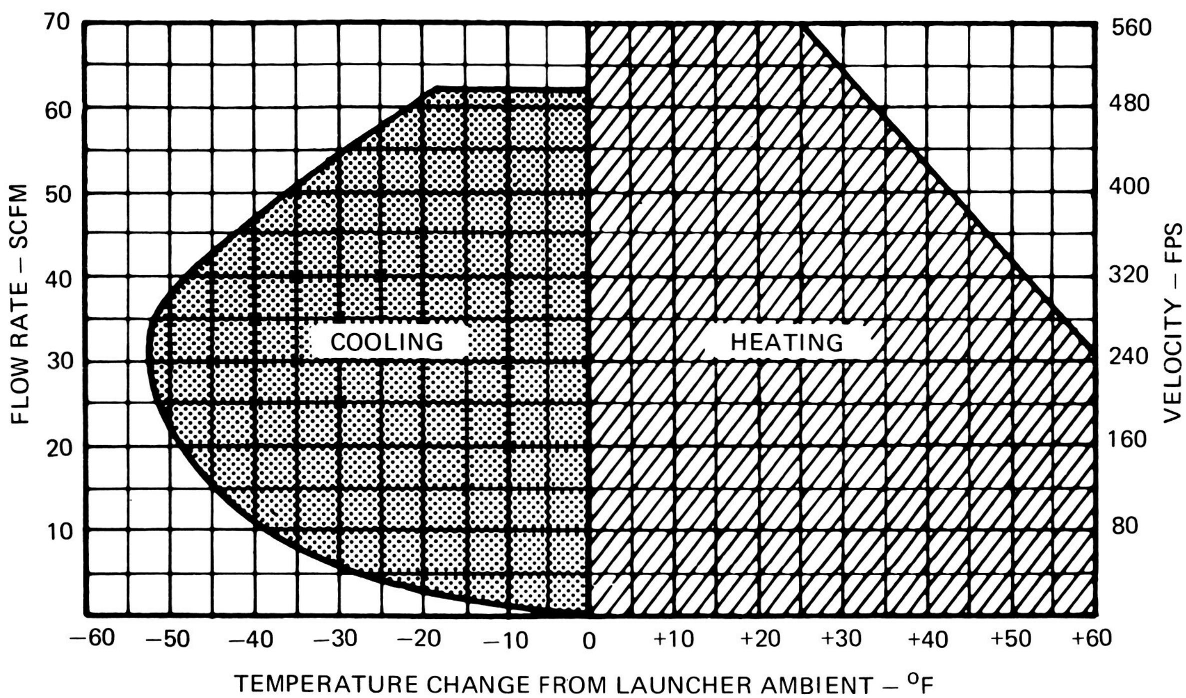

FIGURE IV–23 HEATSHIELD ENVIRONMENT
(HILSCH-TUBE SYSTEM)

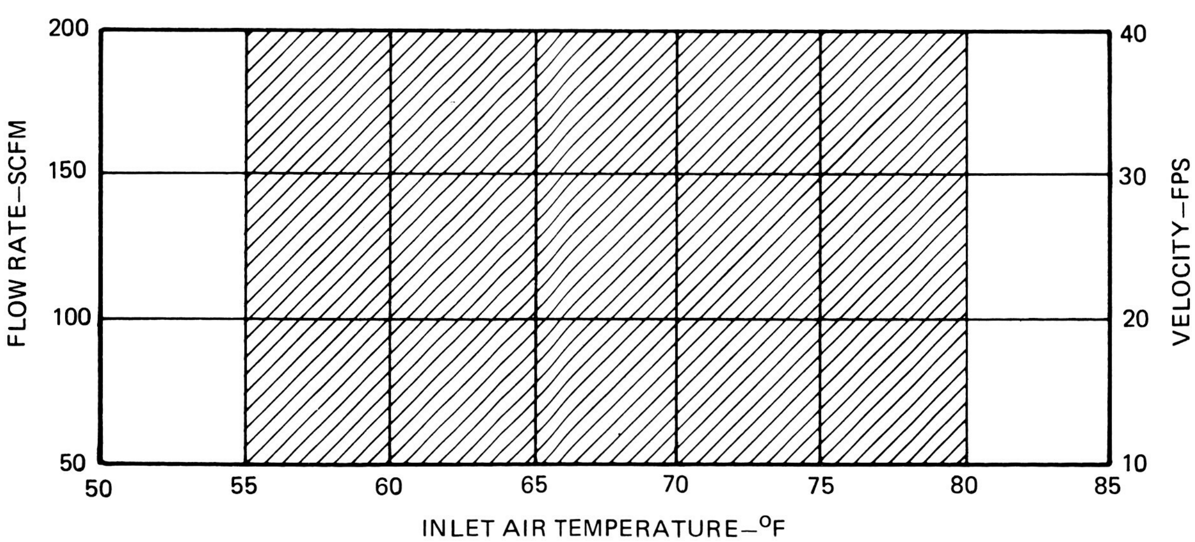

FIGURE IV–23A HEATSHIELD ENVIRONMENT
(HIGH VOLUME SYSTEM)

2.2 LAUNCHER DESCRIPTION

The standard launcher (see Figure IV-24) is a hinged structure with a rotating base. The launcher can be used in a horizontal position for launch preparations and then, with shelter rolled back, elevated to the vertical position for launch. The launcher can be rotated on its base to the required launch azimuth. All electrical connections to the vehicle are of the fly-away type, disconnecting at first motion of the vehicle. Electrical connections to the Payload may be either fly-away type or be command ejected prior to lift-off.

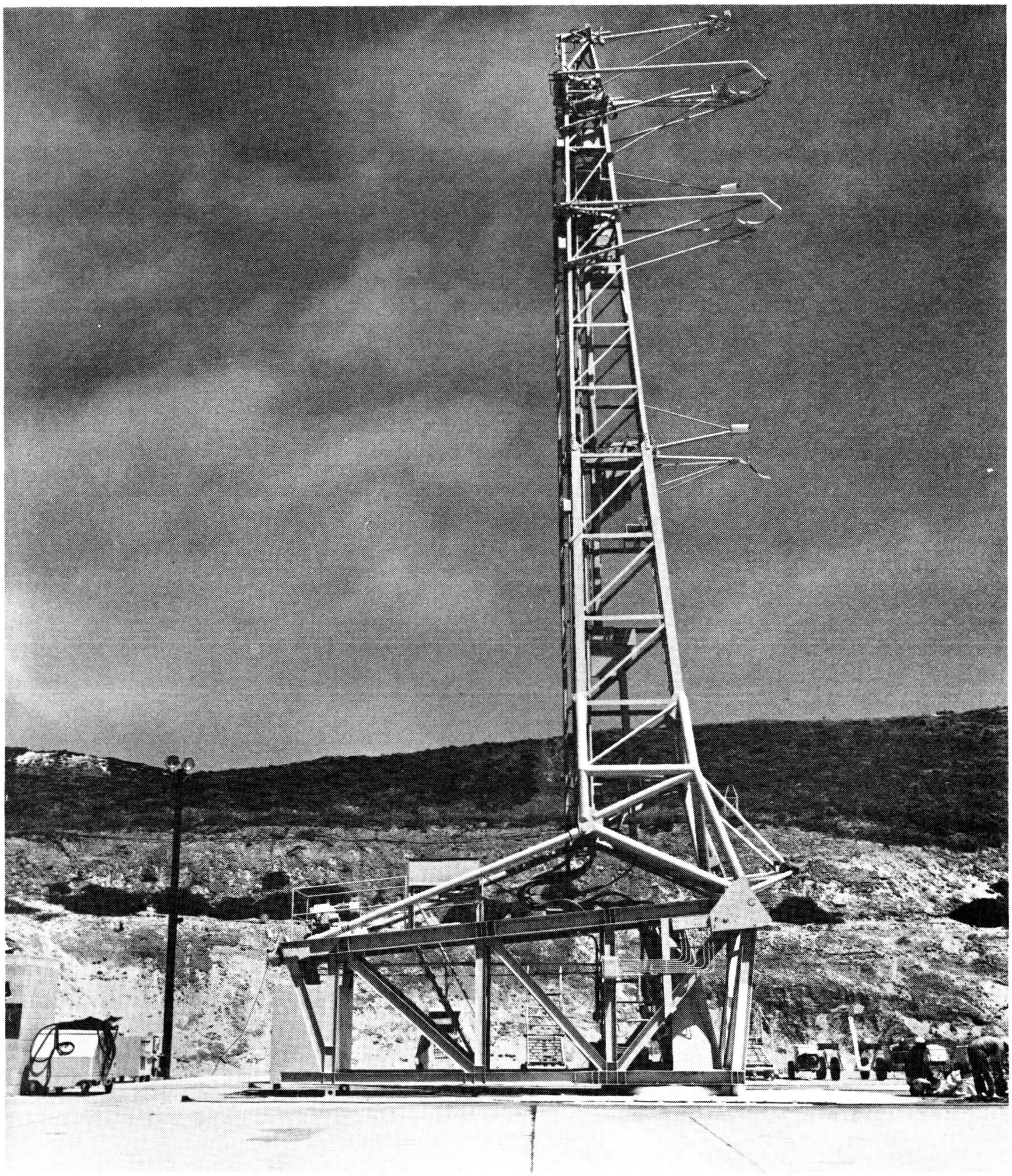

FIGURE IV–24 SCOUT LAUNCHER – VAFB

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2.3 LAUNCH COMPLEX CABLING

A simplified diagram of the Launch Complex cabling system is shown in Figure IV-25. As shown in the simplified diagram, the payload electrical system is continuous through a series of (J) boxes and terminal room between blockhouse and launcher. The launcher electrical system terminates in an umbilical cable arm three feet from the payload to allow for the variance in payload umbilical requirements. The payload umbilical cable is fabricated, tailored to a specific payload, and is furnished with the Scout vehicle.

Cabling schematics for the launch complex at each site are shown in Figure IV-26, Sheets 1, 2 and 3.

Cabling termination points for each payload are defined by the Ground Support Equipment-Electrical Interface Drawing.

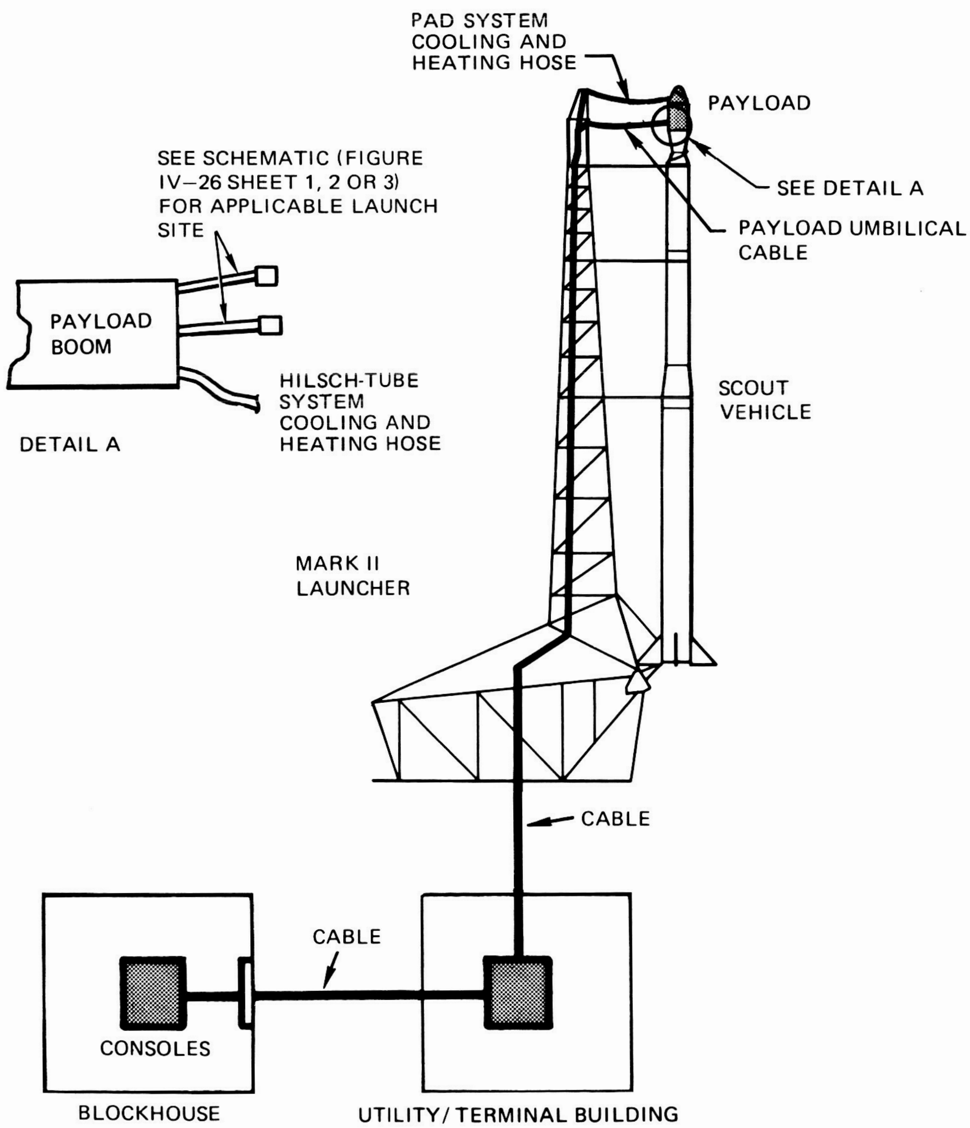

FIGURE IV–25 BLOCKHOUSE TO PAYLOAD CABLING SIMPLIFIED DIAGRAM

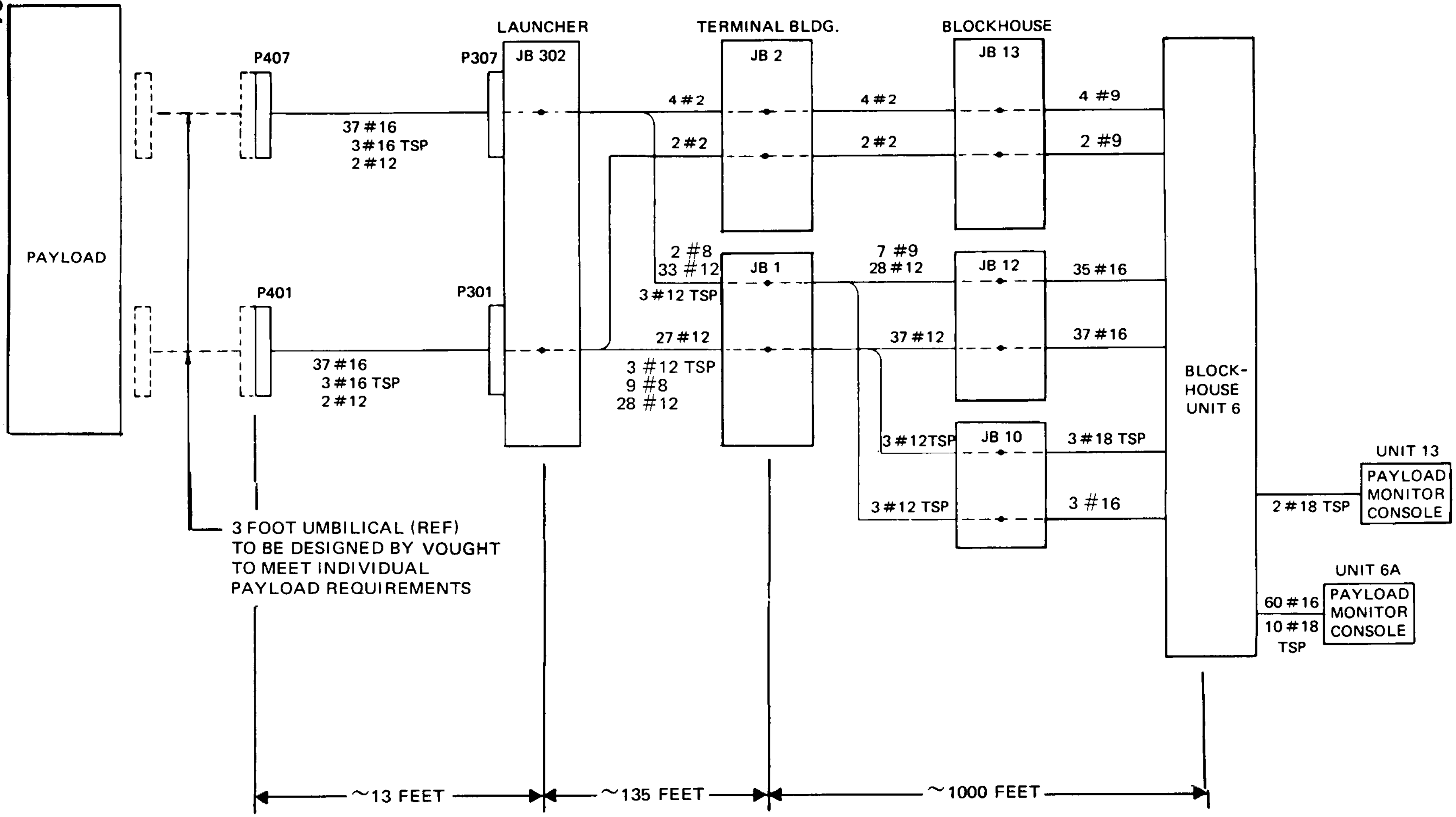

FIGURE IV–26 LAUNCH COMPLEX INTERCONNECTING CABLE SCHEMATIC (SHEET 1 – WFC)

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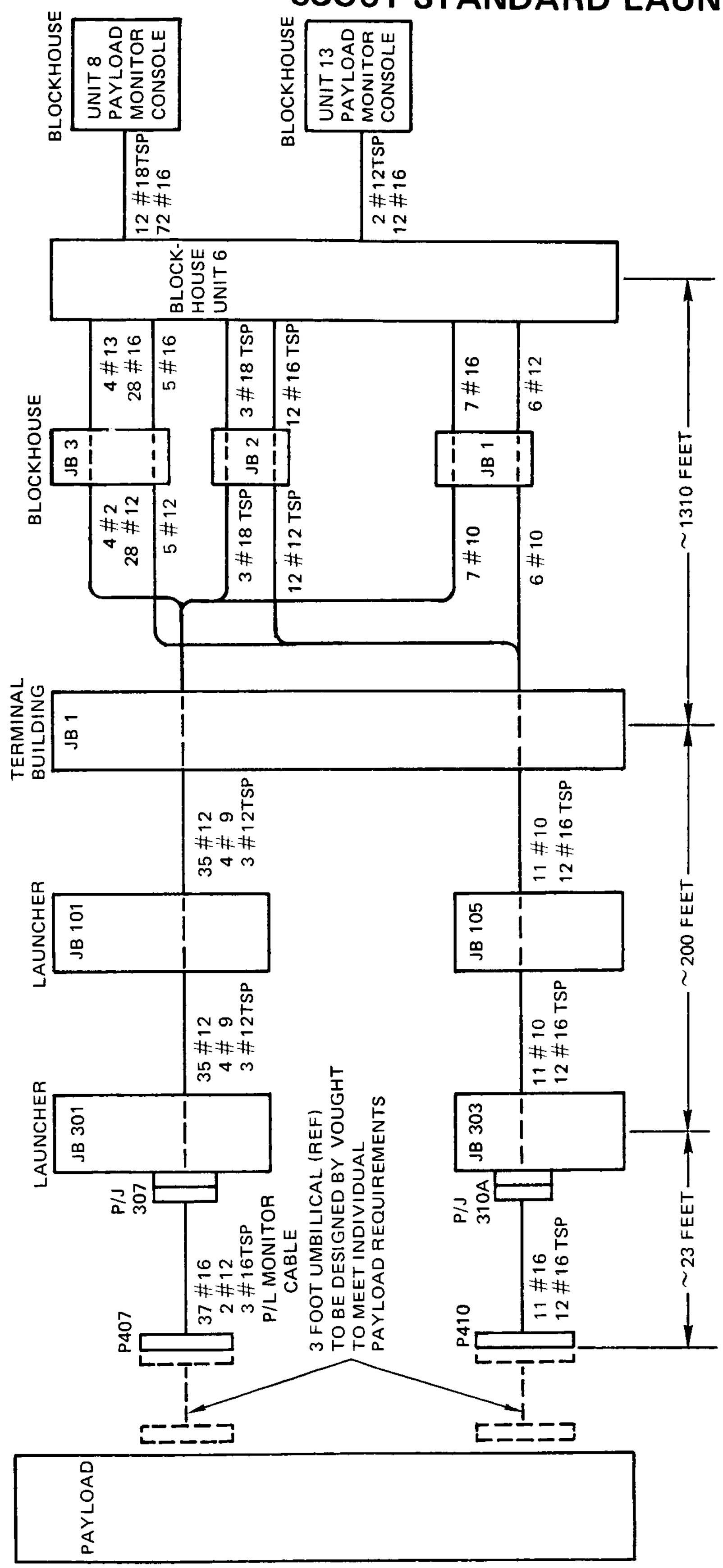

FIGURE IV–26 LAUNCH COMPLEX INTERCONNECTING CABLE SCHEMATIC (SHEET 2 – VAFB)

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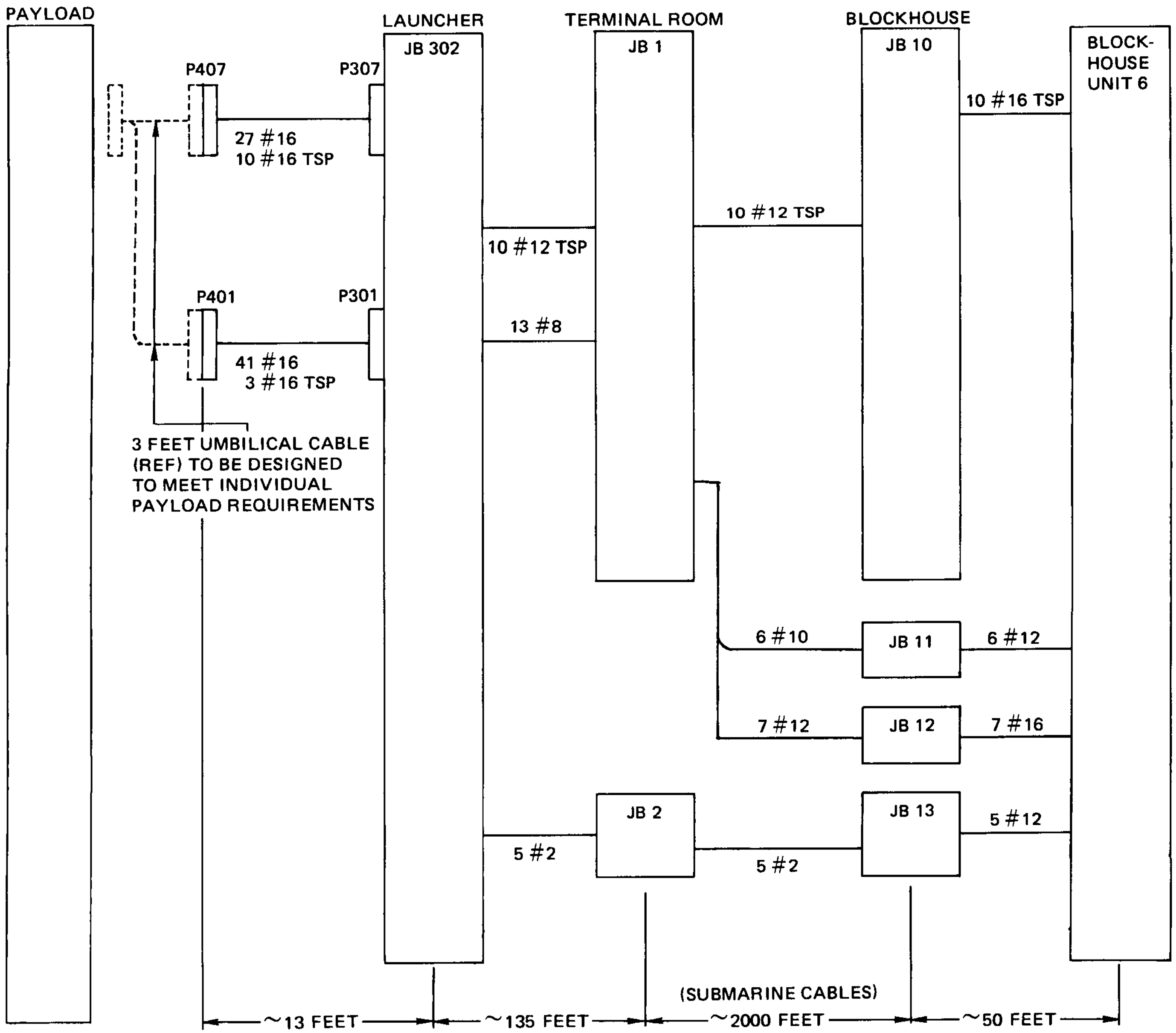

1. ADDITIONAL WIRES MAY BE SUPPLIED AT THE LAUNCHER OR BLOCK-HOUSE

2. CABLES BETWEEN THE TERMINAL ROOM AND BLOCKHOUSE JUNCTION BOXES ARE PERMANENT SUBMARINE CABLES.

FIGURE IV–26 LAUNCH COMPLEX INTERCONNECTING CABLE SCHEMATIC (SHEET 3 – SAN MARCO)

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2.4 BLOCKHOUSE

Vehicle launch control is performed from the Blockhouse by the launch team operating under the direction of a Test Director. The Scout launch control consoles are arranged in the Blockhouse in two rows with the operator consoles making up the front row and the supervisor consoles making up the back row.

Specific consoles are allocated for Payload Agency use. The Payload Agency's checkout equipment used in the Blockhouse shall be on panels which will be installed in these console control units. The console panel envelope available for Payload Agency use is defined in this section.

Details of the San Marco Range Blockhouse are not included in this manual. (Refer to San Marco Range User's Manual.) The general arrangement of the San Marco Blockhouse consoles is similar and is functionally identical.

2.4.1 Wallops Flight Center Blockhouse

The Wallops Flight Center Blockhouse (See Figure IV-27) is a hemispherical-type two story, concrete structure, that houses the launch control and monitoring equipment used during Scout launches. A/C power, 110V 60 hertz outlets, and an antenna facility connected to the lower level by four coax cables and 12 conductors supplied from payload patch panels, are available on the roof of the Blockhouse for payload use. This is incidental to the launch operation electrical system and may be utilized by the Payload Agency for satisfaction of open loop R-F payload control requirements, if any. See Figure IV-28 for Blockhouse floor plan. Figure IV-29 illustrates Blockhouse personnel seating arrangements, and details the function of each console.

FIGURE IV–27 SCOUT BLOCKHOUSE – WFC

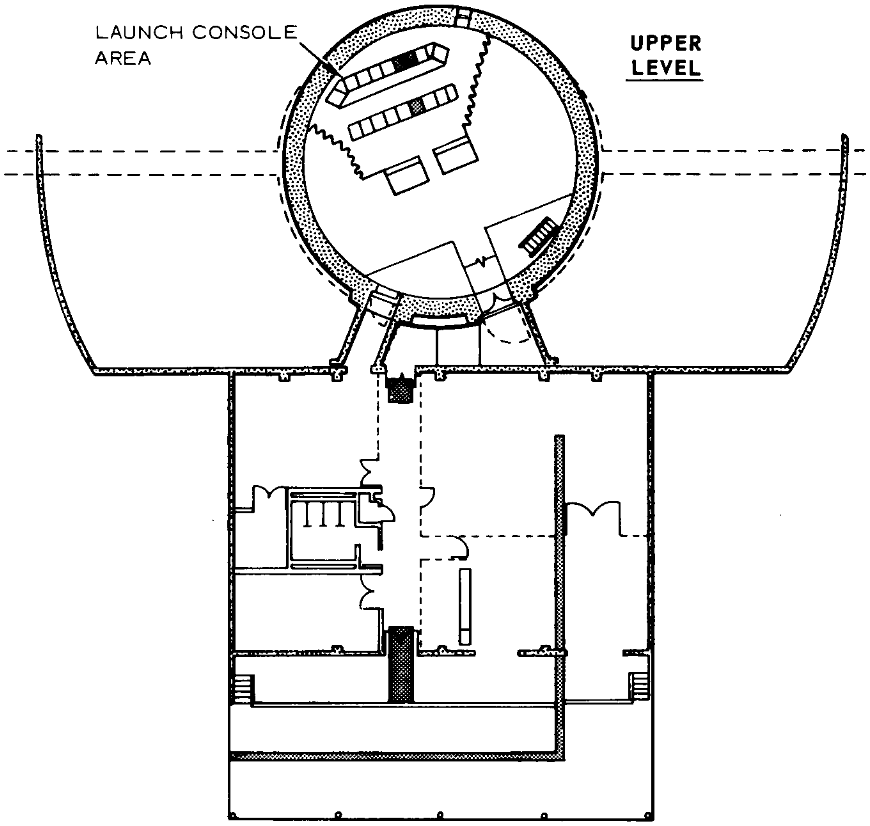

FIGURE IV–28 WFC BLOCKHOUSE FLOOR PLAN

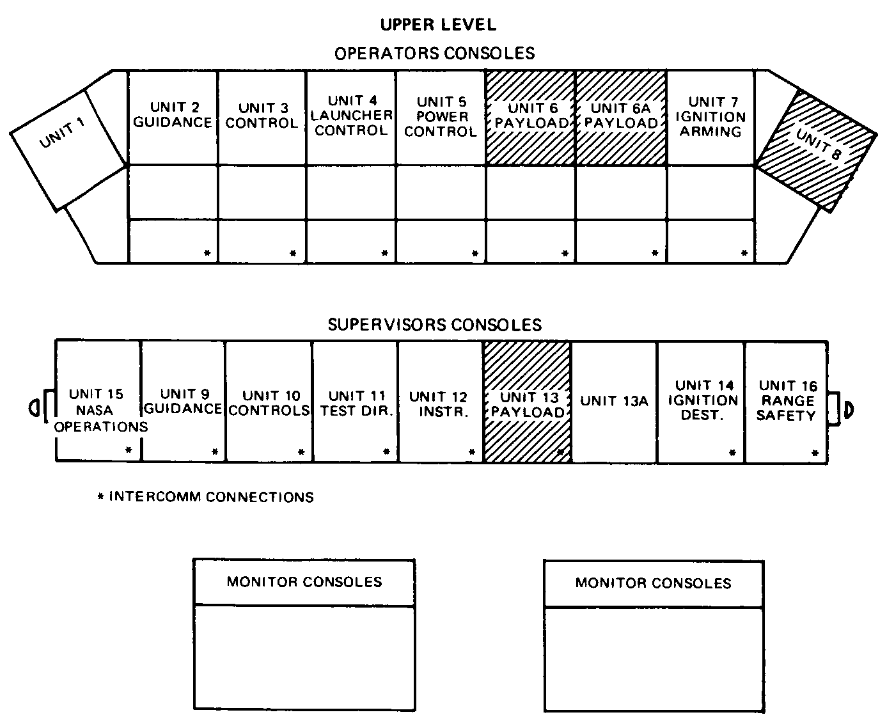

FIGURE IV–29 WFC BLOCKHOUSE PERSONNEL SEATING ARRANGEMENTS AND CONSOLE FUNCTIONS

2.4.1.1 Blockhouse Consoles – WFC

The launch console control units 6, 6A, and 13 are designated for payload use.

The Payload Agency's payload checkout equipment used in the blockhouse shall be on panels which will be installed in console control units 6 and 6A.

Unit 1 –
This console provides a capability to remotely control and monitor the temperature and flow of the payload environmental system air supplied by the ECS umbilical prior to flight. It also provides the capability of monitoring five additional heatshield or payload temperatures.

Unit 4 – Launcher Control Console
This console provides the capability to remotely monitor and control the heatshield inlet temperature of the cooled, or heated, oil-free air supplied by the lower capacity Hilsch-tube system. It also provides an indication of the position (in or out) of the launcher payload umbilical arms.

Unit 6 – Payload Control Console
The vertical panels provide a 19.50 x 19 inch panel space for payload use. The 8.25 x 19 inch panel mounted below the vertical payload panel provides monitor and control of the Fourth Stage Module standard separation system. The sloping panel provides a 17.5 x 19 inch standard EIA panel space for payload use.

Unit 6A – Payload Monitor
This unit provides a 27.50 x 19 inch vertical panel space and a 17.5 x 19 inch sloping panel space for payload use.

Unit 8 – Payload Monitor Console
This unit provides a 45.0 x 19 inch panel space and a 21 x 19 inch panel space which can be made available for payload use.

Unit 13 – Payload Supervisors Console
This console contains two payload monitoring recorders, two 0-1 hour timers, an umbilical INTERLOCK circuit energization and indicator, 8 various bus voltage status indicator lights, and a HOLD switch and PROCEED indicator for Payload use. The recorders are dual pin, Taylor Instrument Corp., recording voltmeters with 0-5VDC range and 0-100 scale. Each recorder has two chart speeds, 4-inches per hour or 1-inch per minute.

Figure IV-30 shows views of typical consoles. Figure IV-31 defines the envelope of the panel space provided in the consoles for Payload Agency use.

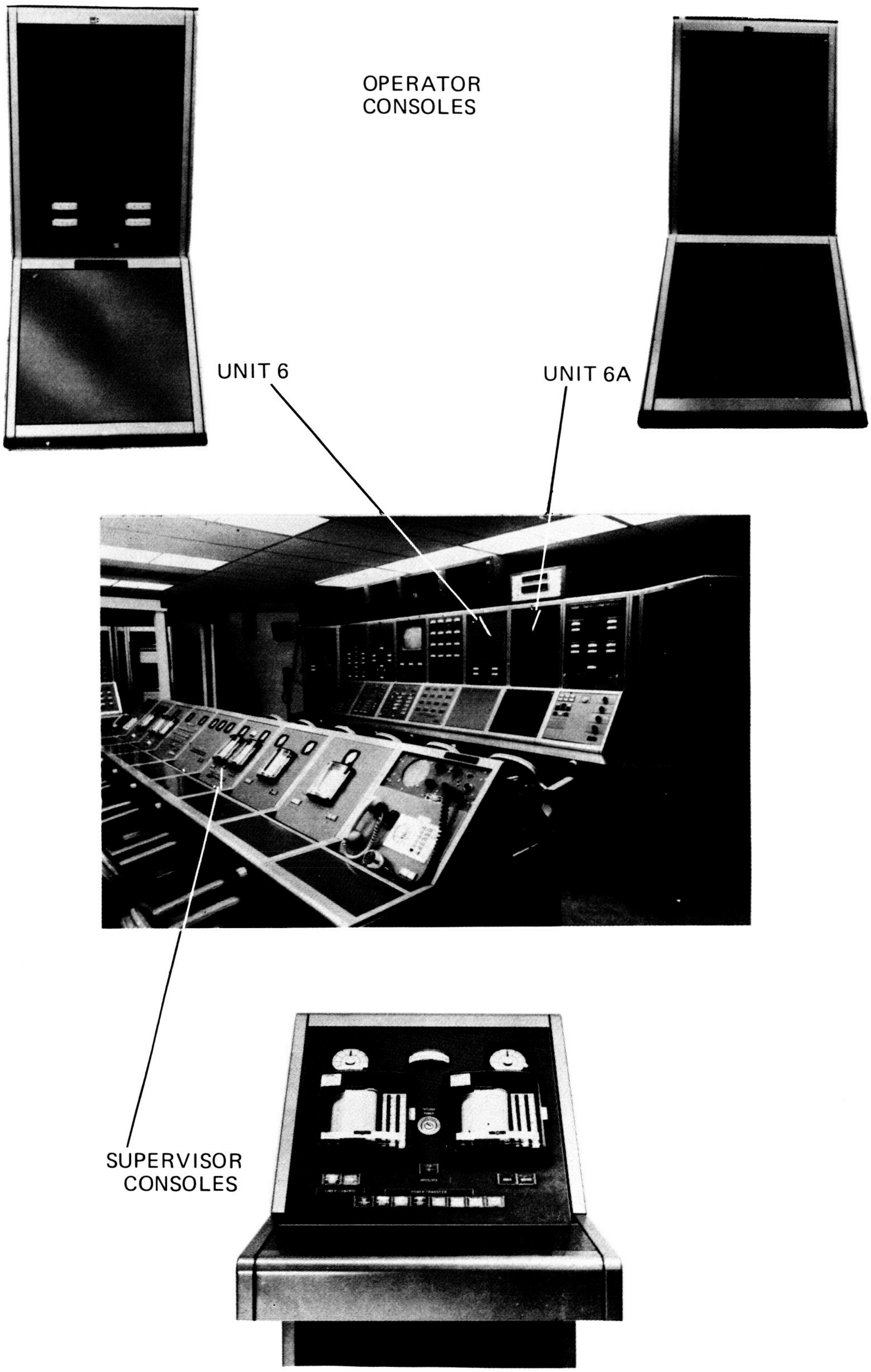

FIGURE IV—30 TYPICAL PAYLOAD CONSOLES — WFC

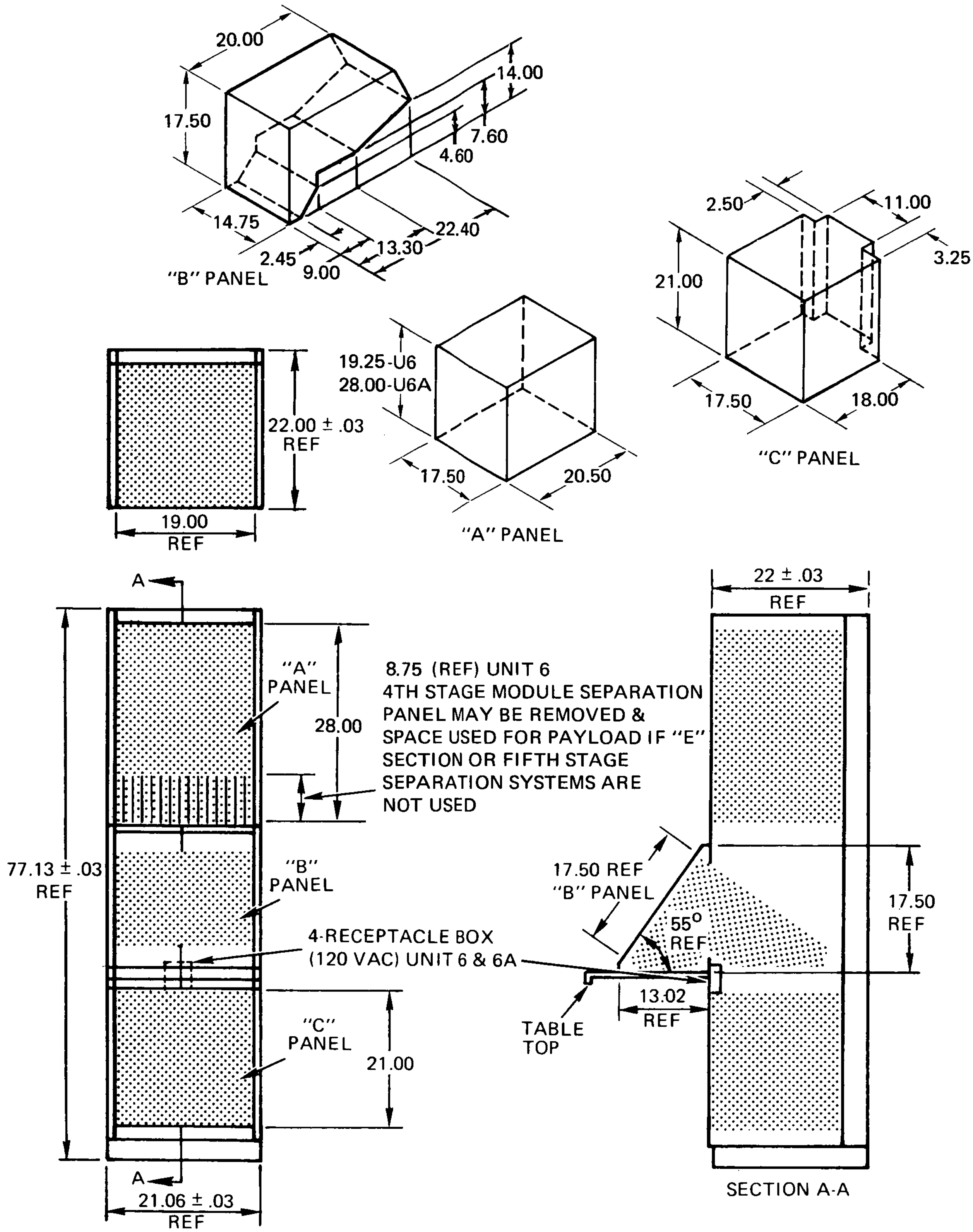

FIGURE IV–31 BLOCKHOUSE CONSOLE UNIT 6 & 6A – WFC PANEL ENVELOPE

2.4.2 Vandenberg Air Force Base Blockhouse

The VAFB Blockhouse (see Figure IV-32) is a 55 feet by 74 feet reinforced concrete structure housing launch and range control equipment. See Figure IV-33 for Blockhouse floor plan. Figure IV-34 illustrates Blockhouse personnel seating arrangements, and details the function of each console.

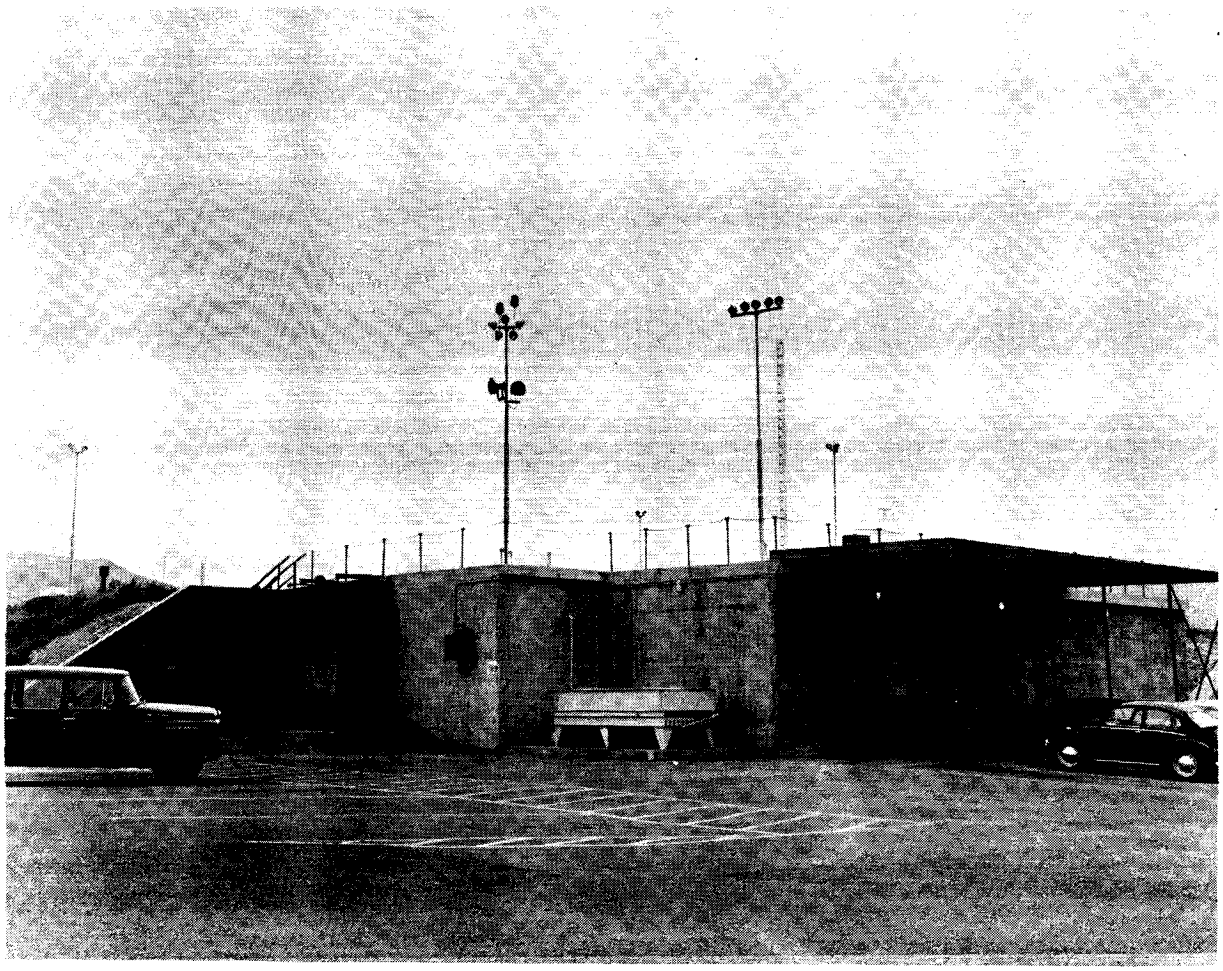

FIGURE IV–32 SCOUT BLOCKHOUSE – VAFB

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SCOUT STANDARD LAUNCH COMPLEX

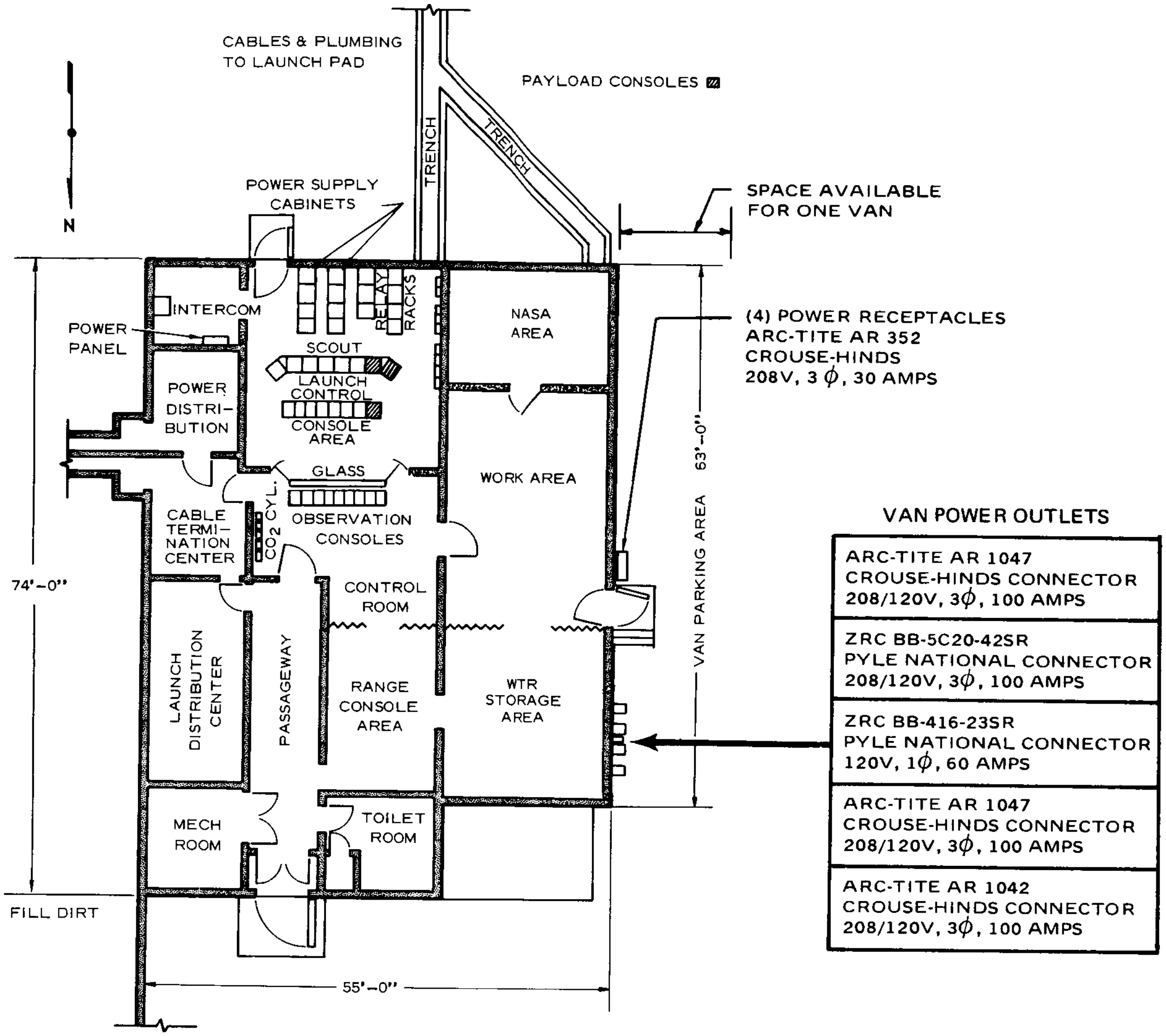


FIGURE IV–33 VAFB BLOCKHOUSE FLOOR PLAN

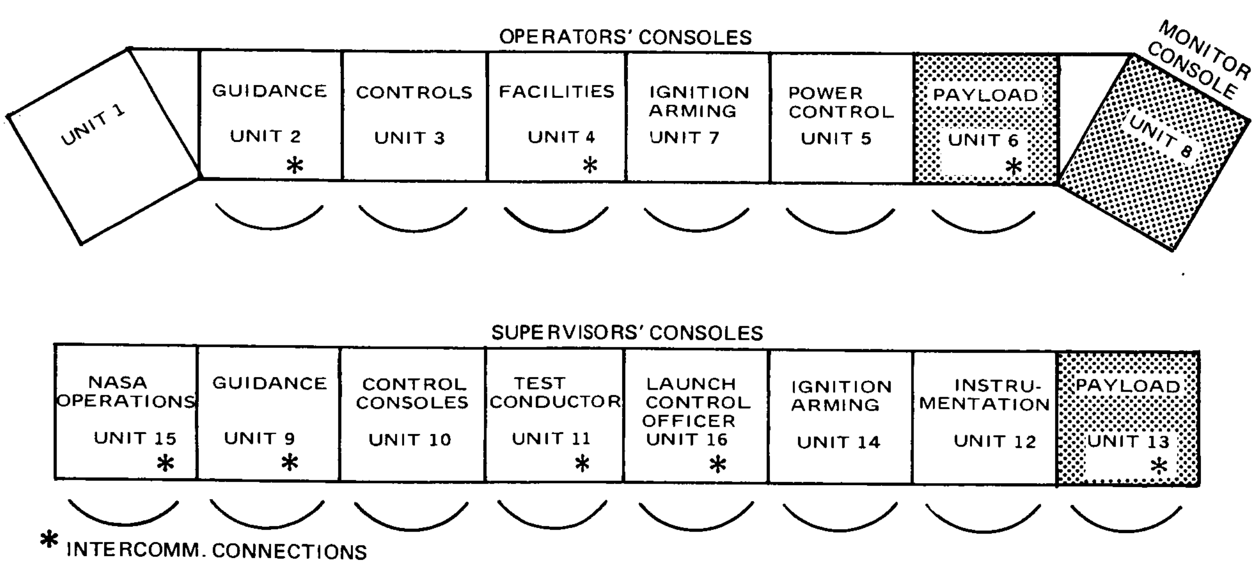


FIGURE IV–34 VAFB BLOCKHOUSE PERSONNEL SEATING ARRANGEMENTS AND CONSOLE FUNCTIONS

2.4.2.1 Blockhouse Consoles —VAFB

The launch console control units 6, 8 and 13 are designated for payload use.

The Payload Agency's payload checkout equipment used in the blockhouse shall be on panels which will be installed in console control units 6 and 8.

Unit 1 —
This console provides a capability to remotely control and monitor the temperature and flow of the payload environmental system air supplied by the ECS umbilical prior to flight. It also provides the capability of monitoring five additional heatshield or payload temperatures.

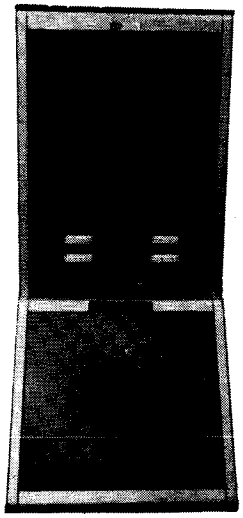
Unit 4 — Launcher Control Console
This console provides the capability to remotely monitor and control the heatshield inlet temperature of the cooled, or heated, oil-free air supplied by the lower capacity Hilsch-tube system. It also provides an indication of the position (in or out) of the launcher payload umbilical arms.

Unit 6 — Payload Control Console
The vertical panels provide a 19.50 x 19 inch panel space for payload use. The 8.25 x 19 inch panel mounted below the vertical payload panel provides monitor and control of the Fourth Stage Module standard separation system. The sloping panel provides a 17.5 x 19 inch standard EIA panel space for payload use.

Unit 8 — Payload Monitor Console
This unit provides a 45.0 x 19 inch panel space and a 21 x 19 inch panel space for payload use.

Unit 13 — Payload Supervisors Console
This console contains a payload monitoring recorder, a HOLD switch and PROCEED indicator, master intercommunication station, and telephone connection for payload use. The recorder is a dual pin, Taylor Instrument Corp., recording voltmeter with a 0-5VDC range and a 0-40000 scale. This recorder has two chart speeds, 4-inches per hour or 1-inch per minute.

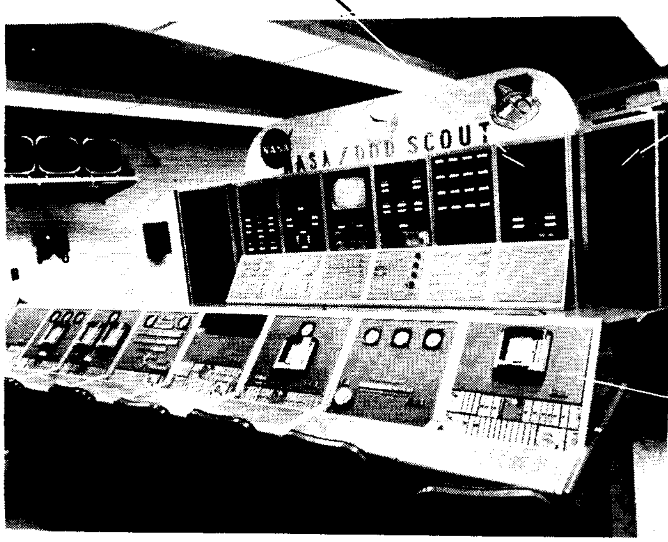
See Figure IV-35 for blockhouse consoles and typical payload consoles. Figure IV-36 defines the envelope of the panel space provided in the consoles for Payload Agency use.



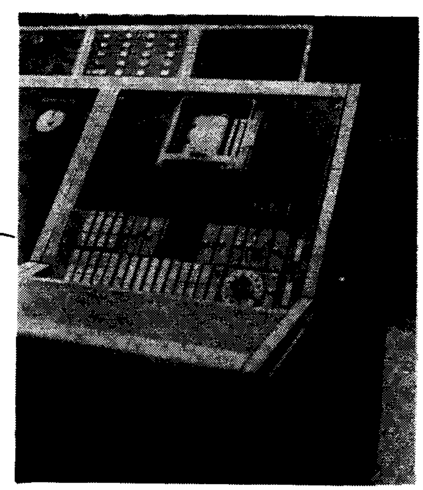
OPERATORS
CONSOLES

UNIT 6

UNIT 8



BLOCKHOUSE INTERIOR
CONSOLE AREA



SUPERVISORS
MONITOR
CONSOLE
UNIT NO. 13

FIGURE IV–35 TYPICAL PAYLOAD CONSOLES – VAFB

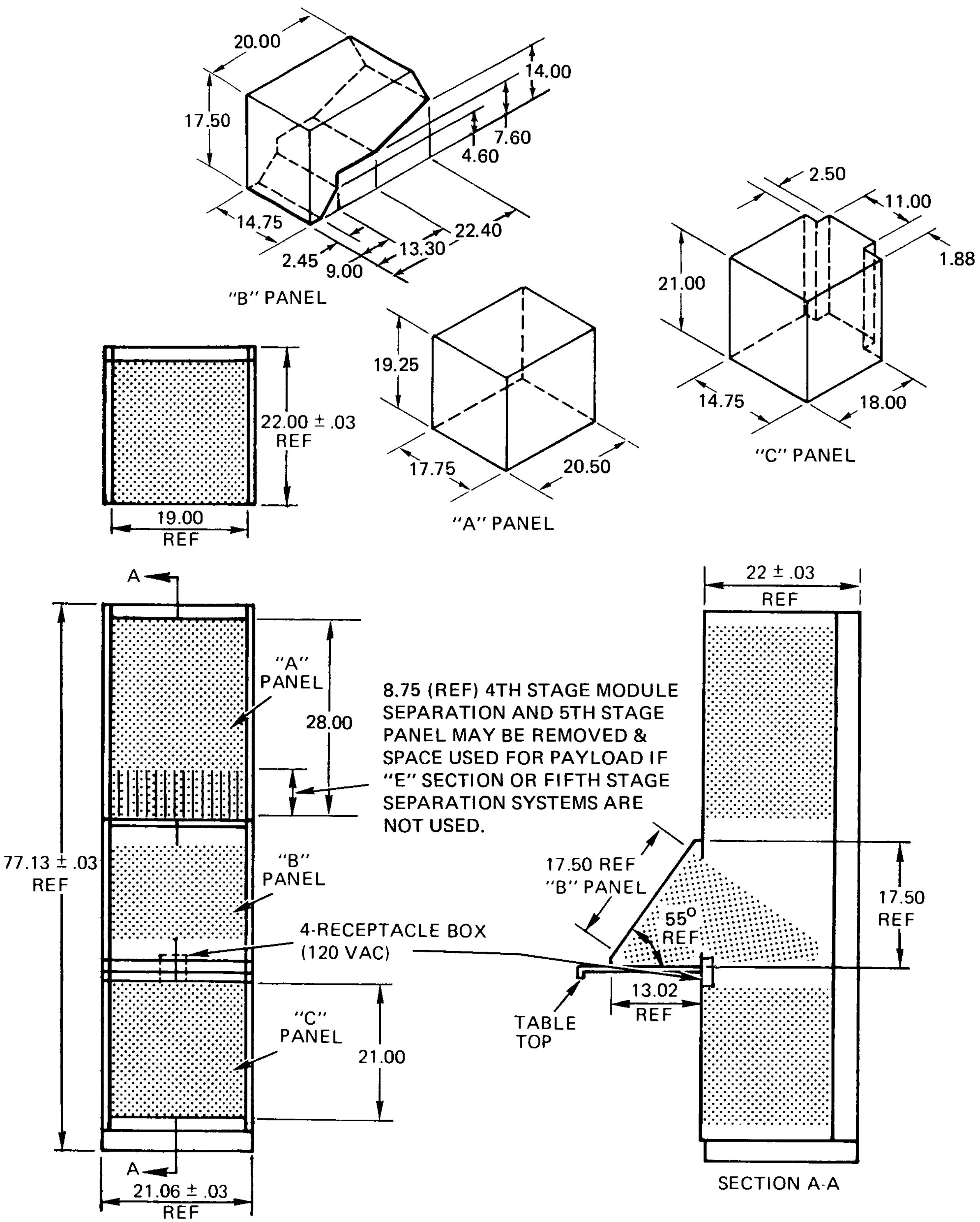


FIGURE IV–36 BLOCKHOUSE CONSOLE – VAFB PANEL ENVELOPE (SHEET 1 – UNIT 6)

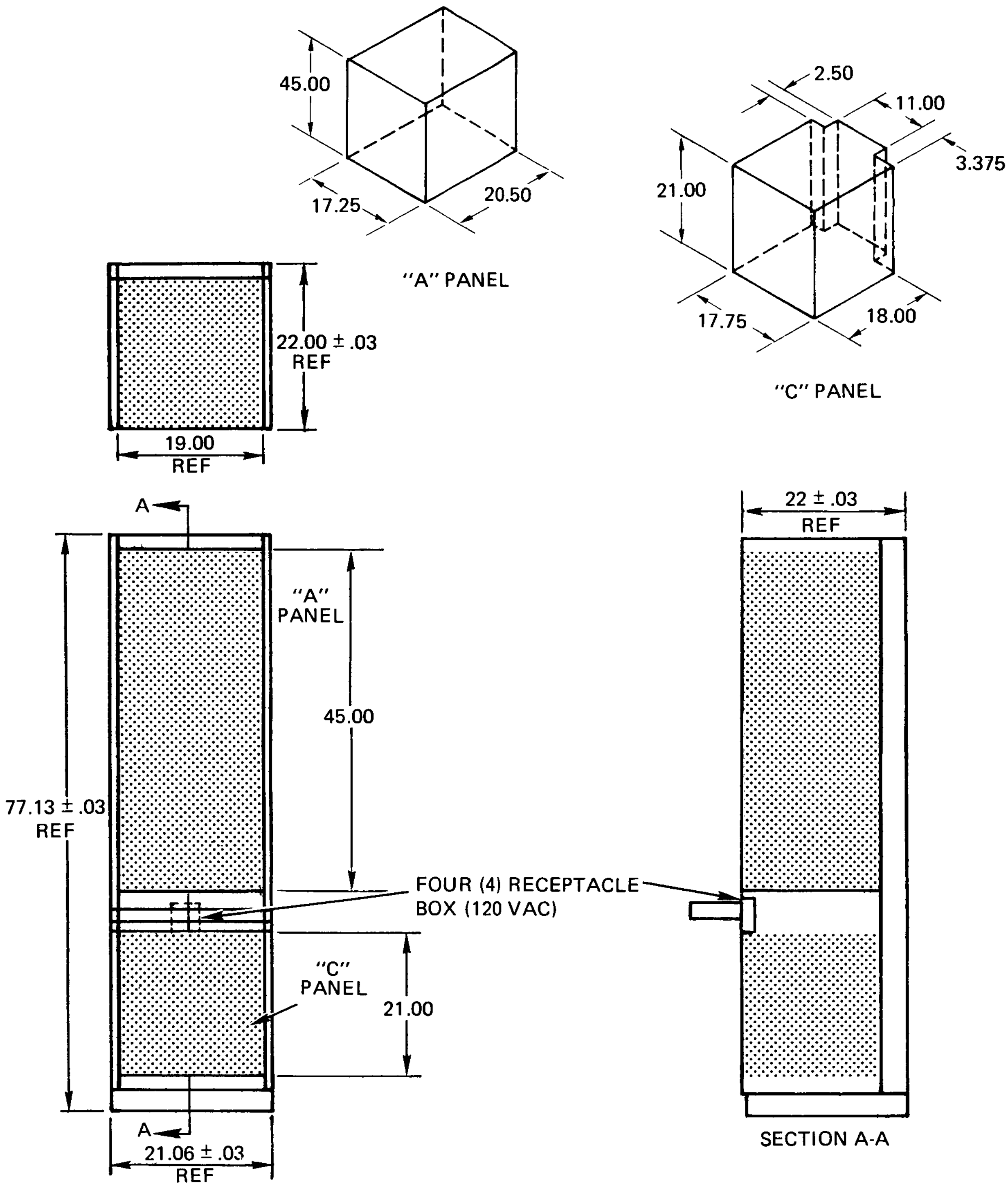


FIGURE IV–36 BLOCKHOUSE CONSOLE – VAFB PANEL ENVELOPE
(SHEET 2 – UNIT 8)

3.0 OPERATIONS

The launch operations are described in this section. The purpose is to aid the Payload Agency in planning and integrating the payload into the Scout launch operations and schedules.

The events and times described herein are based on a typical payload. The Scout vehicle, being operational has sharply defined checkout and operational requirements which are scheduled in an optimum sequence. However, scheduling of the fourth stage build-up, dynamic balancing, RFI, etc. may vary to meet the requirements of the payload. The vehicle operational crew will make every effort to comply with any payload operational requirement that does not compromise vehicle performance or personnel safety.

As soon as sufficient information becomes available, the Vought Corporation Field Team Manager at Wallops Flight Center or at Vandenberg Air Force Base, will prepare a work schedule for each vehicle defining operations from uncrating vehicle sections in Receiving Area to launch of the vehicle at the pad. Work schedules at San Marco are prepared by the CRA launch team. While all schedules are not alike, the average Scout vehicle processing is similar due to the use of Standard Procedures.

3.1 DYNAMIC BALANCING OPERATIONS

Fourth stage buildup and spin balancing procedures begin with receipt of the flight "D" section, fourth stage motor, payload separation system and associated hardware at the dynamic balancing facility after completion of receiving inspection and component testing. Task performance begins with preparation of the balance machine followed by fourth stage buildup, alignment checks, concentricity checks and the dynamic balancing operation.

The Payload Agency is required to provide handling equipment for both vertical and horizontal assembly and disassembly of the payload to the fourth stage motor.

The minimum spin rate to be used during dynamic balancing is the in-flight predicted +3 sigma spin rate. Using this spin rate ensures structural adequacy and increases the accuracy of the balancing procedure.

There are three methods by which payload buildup, dynamic balancing and fourth stage handling may be accomplished. These methods are shown pictorially in Figure IV—37.

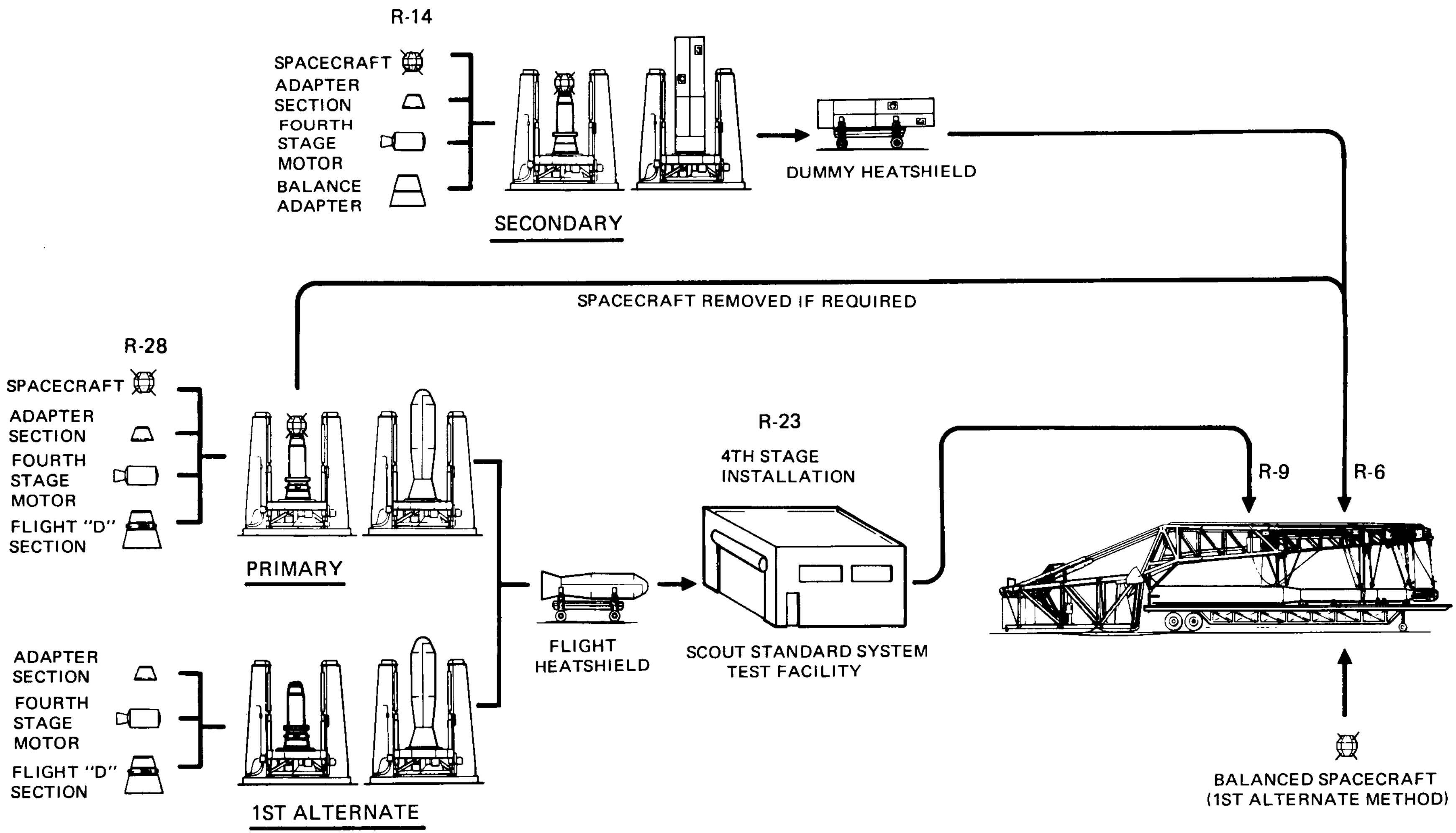


FIGURE IV–37 PAYLOAD FIELD FLOW PROCESSING PLAN

In order of preference, these methods are:

A. Primary Method – The primary method for dynamic balancing utilizes the vehicle flight "D" section, fourth stage motor, payload separation system, associated hardware and payload. This method requires the balance operation to begin at R-28, prior to vehicle systems test. However, if subsequent payload tests are required, the payload may be removed following the balance operation and rejoin the vehicle on the pad at approximately R-6.

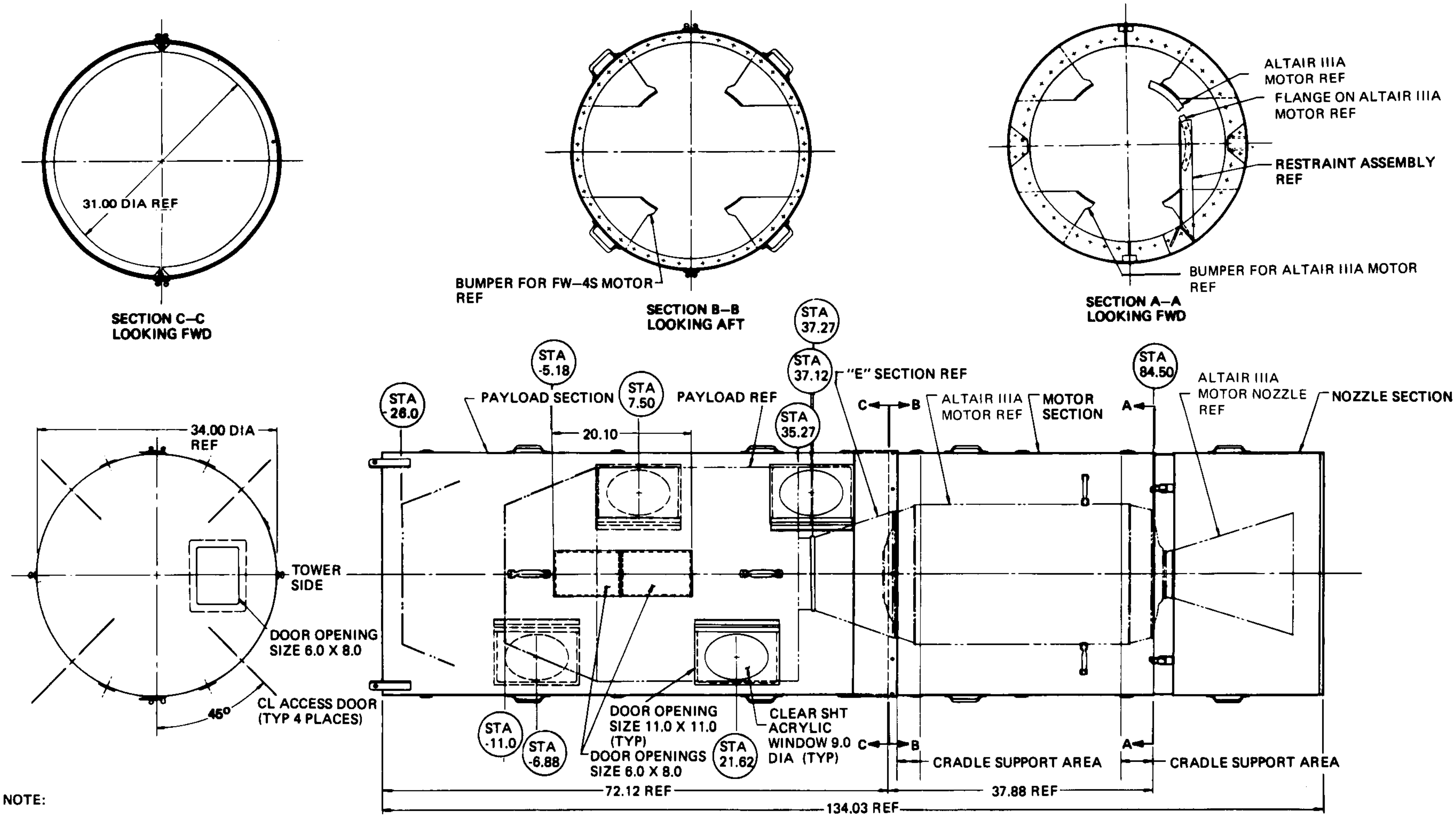
B. Alternate to Primary Method – This method utilizes the same component listed above with the exception of the payload. It allows the payload to be balanced independently to the limits specified in Chapter V and to mate with the vehicle at the launch pad at approximately R-6.

C. Secondary method – The secondary method utilizes a balance adapter (in lieu of the flight "D" section) fourth stage motor, payload separation system, associated hardware and payload. Spin balancing by this method begins at R-14 and permits simultaneous vehicle system testing.

When the primary balance procedure is used it is desirable to have the spacecraft balanced with the fourth stage to minimize tip-off and to assure maximum accuracy. In some cases where it is not practical to balance with the fourth stage it will be necessary to provide assurances of proper spacecraft balance when mated to the vehicle. This assurance can take the form of a check spacecraft balance in the field using Scout dynamic balance facility. If the conducting of this check balance is not technically possible alternate assurances must be provided at the discretion of the Scout Project Office working through the Mission Working Group. The detailed schedule and the method used to perform these checks are topics of discussion for the Mission Working Group.

3.1.1 Dummy Heatshield

When the payload-fourth stage assembly is balanced by the secondary balance method, the use of a dummy heatshield is required (Figures IV-38 and IV-39). This heatshield is used to provide protection to the payload-fourth stage assembly during transport from the spin balance facility to the pad. It is available in two diameters; corresponding to the diameters of the flight heatshields. Each heatshield is composed of: two forward sections, with two transparent access doors in each section; two center sections, with four motor supports in each section; and a section for protection of the motor nozzle. The payload is not required to provide support areas as all loads are transmitted to the motor case.



NOTE:

BASIC STRUCTURE AND OUTER SKIN OF DUMMY HEAT SHIELD ASSEMBLY FABRICATED FROM ALUMINUM

FIGURE IV–38 34 INCH DIAMETER DUMMY HEATSHIELD ASSEMBLY

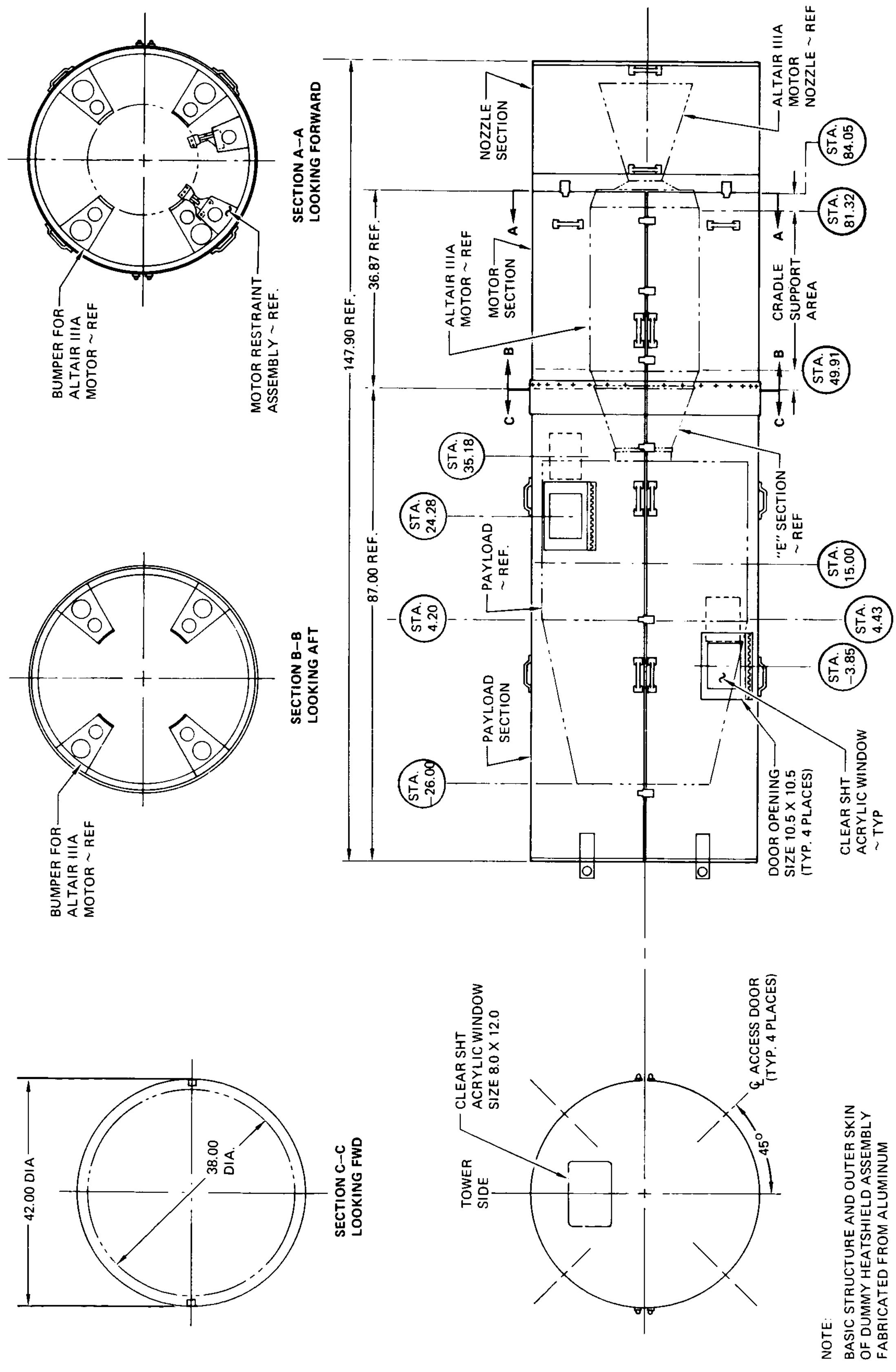


FIGURE IV–39 42 INCH DIAMETER DUMMY HEATSHIELD ASSEMBLY

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3.2 TYPICAL PRE-PAD OPERATIONS

The processing dates given in this section are based on the Primary Method and are the latest possible to meet the scheduled launch date.

Launch site operations begin approximately 36 working days before launch (R-36) when the rocket motors are received, uncrated and inspected. Each motor takes 2 to 5 days to complete.

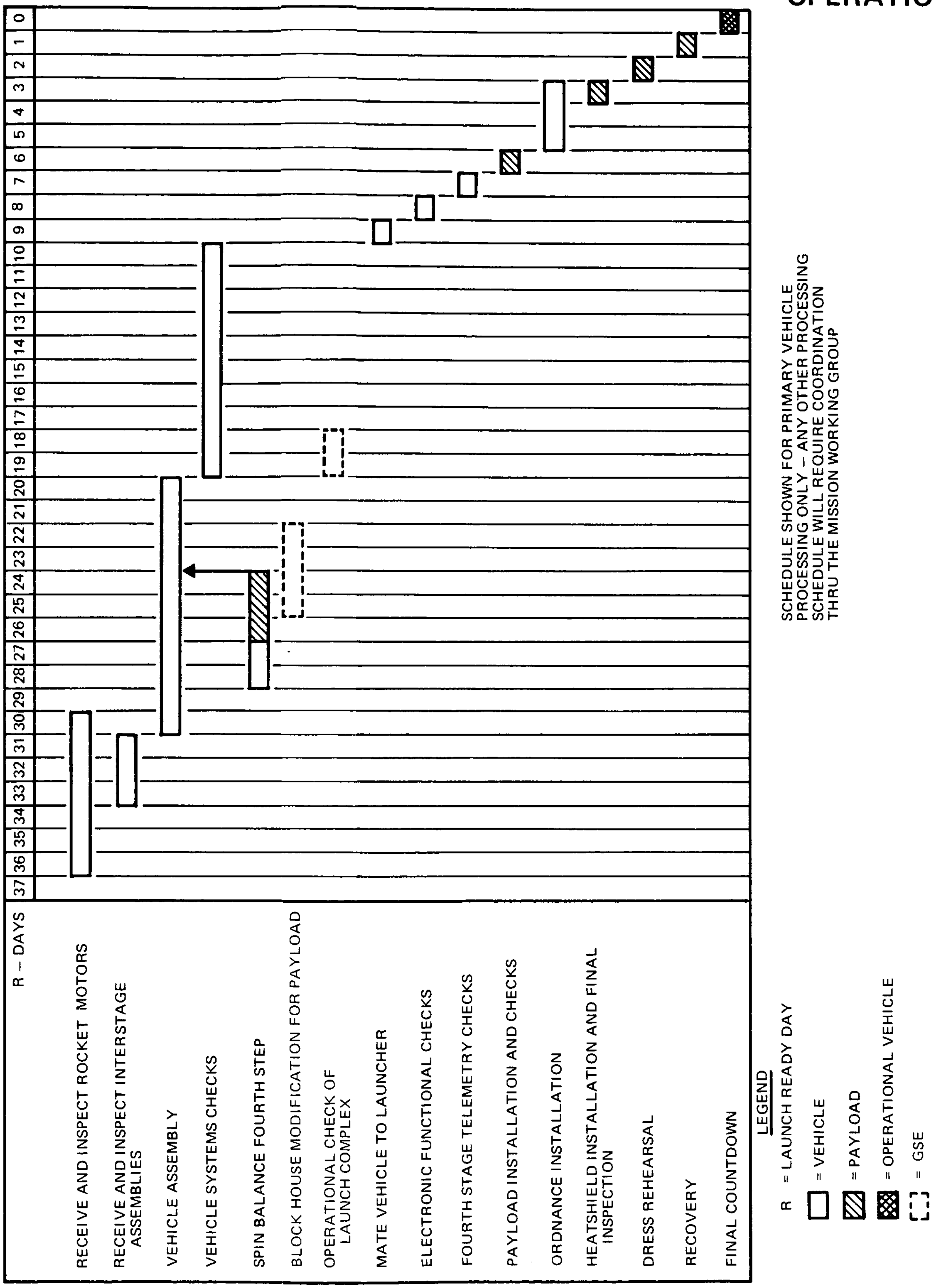
At R-33, the interstage assemblies are received, uncrated, and inspected. This operation takes approximately 3 days to complete.

At R-30, the vehicle interstage and rocket motor assembly is started. The base A and first stage motor are assembled on the transporter. B section and second stage motor, C section and third stage motor assemblies are assembled on dollies and then transferred to the transporter for vehicle assembly. The balanced D section and fourth stage are installed on the vehicle at R-23. Approximately 11 days are required to assemble the vehicle.

At R-19, the assembled vehicle systems checks are started, approximately 10 days are required to complete these checks. On R-11, the All Systems and the preliminary RFI checks are performed.

At R-9, the assembled vehicle is mated on the launcher.

Refer to Figure IV-40 for a simplified flow diagram of the vehicle processing and pad operations time schedule.



SCHEDULE SHOWN FOR PRIMARY VEHICLE PROCESSING ONLY – ANY OTHER PROCESSING SCHEDULE WILL REQUIRE COORDINATION THRU THE MISSION WORKING GROUP

FIGURE IV–40 TYPICAL WORK TIME SCHEDULE

3.3 TYPICAL ON-PAD OPERATIONS

Operation at the pad begins with the blockhouse modifications for the payload, and preparation of the launcher. This task, depending upon complexity, takes from 6 to 10 days. Blockhouse modification for the payload should be completed prior to the operational check of the launch complex. At R-19, 2 days are allowed for the operational check of the launch complex. At R-9, the assembled vehicle is mated to the launcher followed by an electronic functional test on R-8. If the payload was removed from the fourth stage following the spin balance operation, it is re-installed at R-6. The payload agency is required to provide handling equipment necessary for payload installation. A bridge crane is provided in the shelter for hoisting and positioning the payload. The ordnance items are installed at R-5 to R-3. A payload umbilical retraction check is optional at R-2 when the dress rehearsal of the operation countdown is performed. Final vehicle preparations are made the day before launch.

TABLE IV–1 TYPICAL ON-PAD OPERATIONS

OPERATION	BEGIN	DELTA TIME
PREPARE LAUNCH COMPLEX	R-29	10 DAYS
BLOCKHOUSE MODIFICATION FOR PAYLOAD	R-25	1 - 4 DAYS
LAUNCH COMPLEX OPERATIONAL CHECK	R-19	2 DAYS
MATE VEHICLE TO LAUNCHER	R-9	1 DAY
ELECTRONIC FUNCTIONAL	R-8	1 DAY
FOURTH STAGE TELEMETRY CHECKS	R-7	1 DAY
PAYLOAD INSTALLATION AND CHECKS	R-6	1 DAY
ORDNANCE INSTALLATION	R-5	3 DAYS
HEATSHIELD INSTALLATION AND FINAL INSPECTION	R-3	1 DAY
DRESS REHEARSAL	R-2	1 DAY
RECOVERY (PREPARE VEHICLE FOR LAUNCH)	R-1	1 DAY
FINAL COUNTDOWN	R-0	

3.3.1 Typical Countdown

Table IV–2 outlines the typical operational countdown.

TABLE IV– 2 TYPICAL OPERATIONAL COUNTDOWN

COUNTDOWN TIME	TASK TITLE	DELTA TIME
T-420	COMMUNICATION CHECKS	5 MIN.
T-415	LAUNCH CONSOLE AND VEHICLE GSE SYSTEM ACTIVATION	25 MIN.
T-390	ELECTRONICS SYSTEMS CHECKOUT	135 MIN.
T-X	SPACECRAFT CHECKOUT	
T-255	REACTION CONTROL SYSTEM FUELING	75 MIN.
T-180	VEHICLE/LAUNCHER SECURING AND ERECTION	75 MIN.
T-105	IGNITION AND DESTRUCT SYSTEMS CHECKOUT	60 MIN.
T-45	COUNTDOWN EVALUATION	15 MIN.
T-30	TERMINAL COUNTDOWN	30 MIN.

T-X: SPACECRAFT FUNCTIONAL CHECKOUT TIME MAY VARY DUE TO DIFFERENT REQUIREMENTS OR DESIGN OF EACH SPACECRAFT AND THEREFORE MAY OCCUR PRIOR TO T-320 MIN. NORMALLY SPACECRAFT CHECKS ARE CONDUCTED CONCURRENT WITH VEHICLE TASKS AND MAY START PRIOR TO VEHICLE TASKS IF REQUIRED.

3.3.2 Heatshield Removal Time Penalties

When in the course of vehicle processing, it becomes necessary to gain access to the payload after the heatshield has been installed, the heatshield can be removed with the following time penalties:

A. Top half only removed and replaced adds approximately one hour plus the time for payload servicing.

B. Entire heatshield removed and replaced – adds approximately six hours plus the time for payload servicing.

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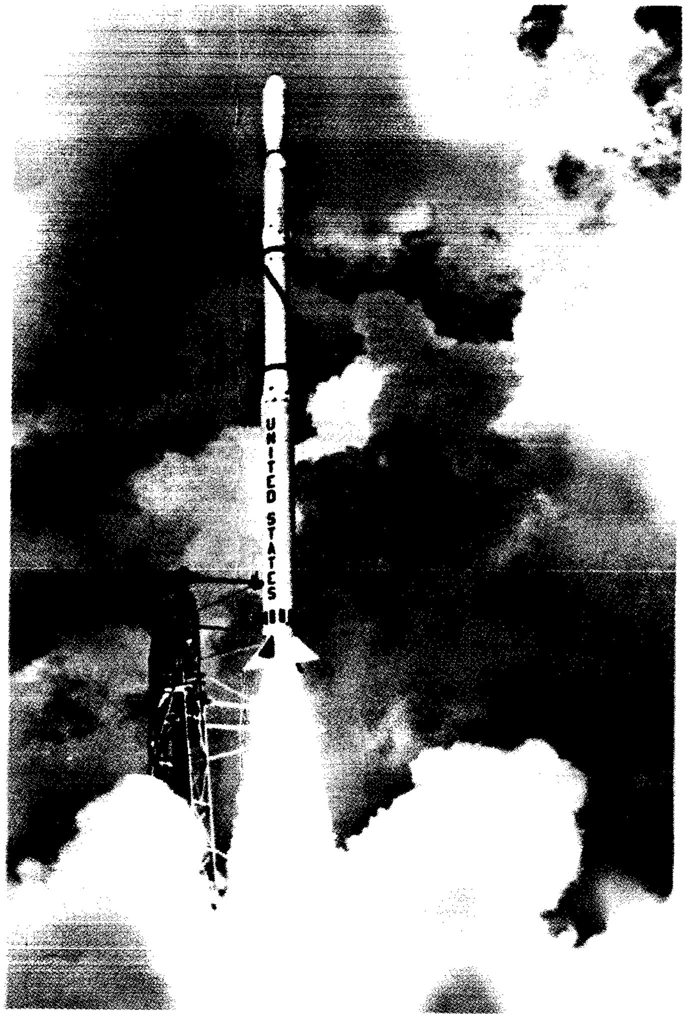

chapter V
DESIGN PARAMETERS

SCOUT
user's manual

1.0 INTRODUCTION

This chapter defines payload design parameters as they pertain to the Scout vehicle and provides the Payload Agency with detail descriptions of payload to vehicle interfaces. Included is information on payload to vehicle adapter/separation systems, payload weight and center-of-gravity location limitations and heatshield payload envelopes.

Payload design criteria are presented which encompass vehicle flight environment, payload test requirements, dynamic balance requirements, electromechanical interfaces, and RFI criteria.

In the design of the payload, there are some constraints that have been imposed by the Ranges. These requirements should be considered during the planning stage of the spacecraft to prevent problems at a later date. The Payload Agency should become familiar with the requirements specified in the Range User's Manual for the applicable range.

1.1 FOURTH STAGE DESCRIPTION

A typical fourth stage with spacecraft installed is shown in Figure V–1.

The Scout fourth stage propulsion unit is the ALTAIR IIIA rocket motor. The solid propellant motor is approximately 20 inches in diameter and has an overall length of 58.43 inches. The motor case is filament wound of fiberglass. The composite propellant grain configuration is a case bonded circular perforation with one transverse slot. The igniter is a flame-producing rocket motor type with dual initiators.

The spacecraft is joined to the fourth stage motor by utilizing one of three basic adapter sections or by mating directly with the motor forward flange utilizing its own adapter section.

The fourth stage is mounted on a large diameter spin bearing attached to the top part of lower transition D section. The unguided fourth stage is spin stabilized prior to separation from the third stage.

Spin-up occurs approximately six seconds prior to fourth stage ignition by a combination of four impulse spin motors. The fourth stage spins clockwise looking forward.

The third and fourth stages are joined by an arrangement of springs held compressed by a flange and a securing clamp. Explosive bolt clamps release the flanges, effecting separation by spring loaded ejection force.

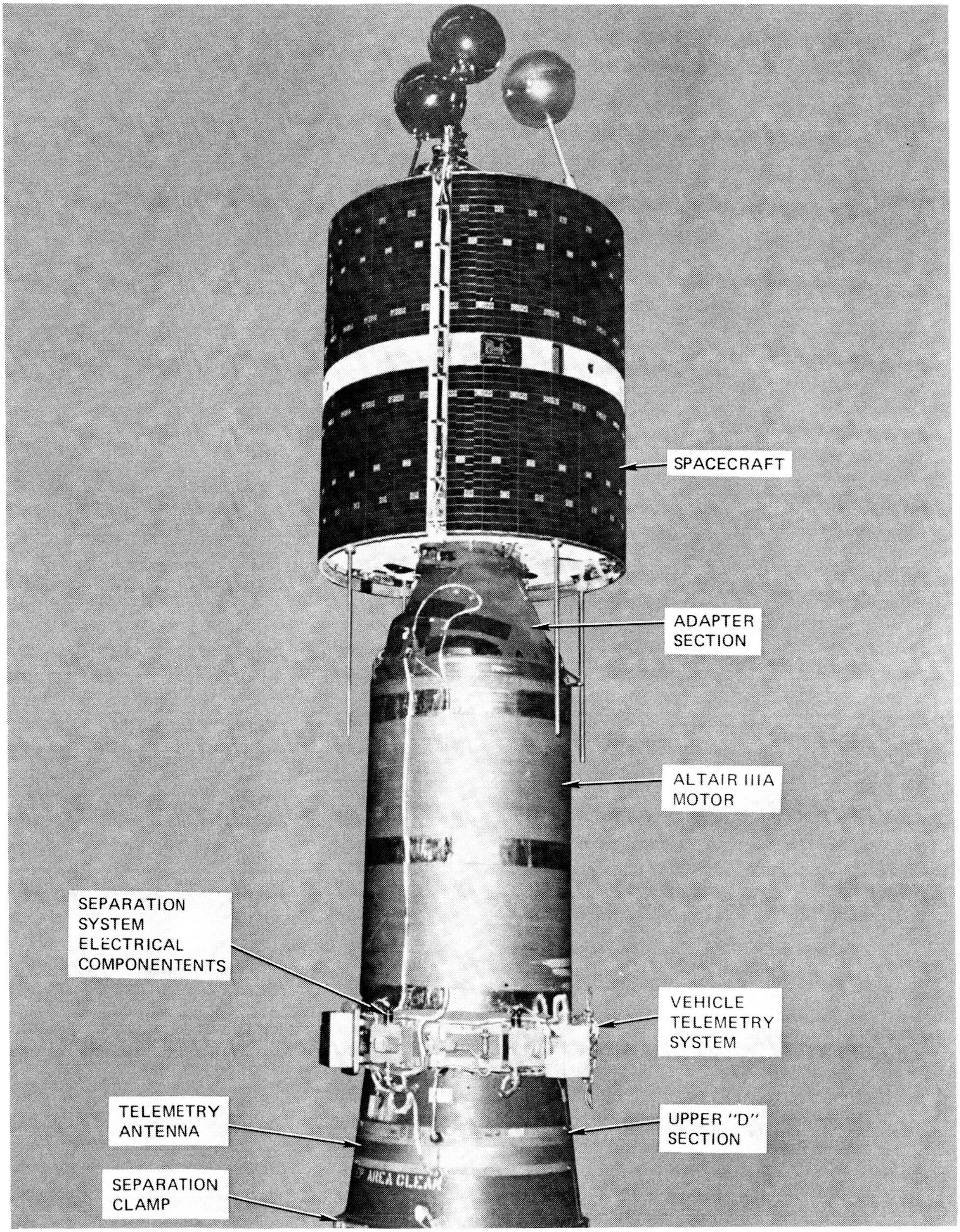

FIGURE V–1 TYPICAL FOURTH STAGE WITH PAYLOAD

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2.0 PAYLOAD TO VEHICLE ADAPTER SECTIONS

Three standard adapter sections, Series 200 "E", Series 25 "E", and Section "EG", are available for mating a payload to the fourth stage motor. The selection of a standard adapter is contingent on payload weight and center-of-gravity location. It should be noted that the payload weight depicted in the adapter capability curves includes only the adapter section and spacecraft weight. The rated load carrying capability is based on structural load tests. Figure V–2 shows the payload weight capability for each standard adapter when the payload center-of-gravity is located at vehicle station 24. The figure provides a quick reference for the initial selection of a standard adapter. The payload weight capabilities of the adapter sections at other center-of-gravity locations are shown in Figure V–3.

A Payload Agency may furnish its own adapter section and attach directly to the final stage motor forward flange. The advantages of using an existing standard adapter becomes apparent, however, when development time and costs associated with a new design are considered. The allowable payload weight in relation to center-of-gravity for Payload Agency furnished adapters is also shown in Figure V-3.

A standard adapter section, "G", is available for mating payloads to the fifth stage motor. The payload to adapter mechanical interface for the "G" section is similar to the other standard adapters. Data on this adapter is not included in this manual but may be provided on request.

The basic adapter assembly consists of a conical adapter, a payload support ring and a payload separation clamp. The adapter base is bolted directly to the final stage motor forward flange. The payload support ring provides threaded holes for attachment to the payload and machined surfaces for mating to top of adapter. The payload separation clamp is a two piece assembly. When bolted together, this clamp holds payload support ring and adapter together. The clamp configuration allows removal of the payload from the vehicle.

The standard adapter sections are designed to provide a separation system when required and to satisfy a mandatory requirement for access to the final stage motor igniters. In general, the design of the standard adapter/separation system is basically the same for all sections.

A Separation Systems Test Planning Manual is available to aid the Payload Agencies in the spacecraft development testing. The manual provides the correct assembly and test procedures for the Scout standard Adapter/Separation Systems.

2.1 PAYLOAD SEPARATION SYSTEM

The separation system is composed of separation springs added to the basic adapter and pyrotechnic devices on the payload separation clamp. Restraint bands and cables are added to control the payload separation clamp

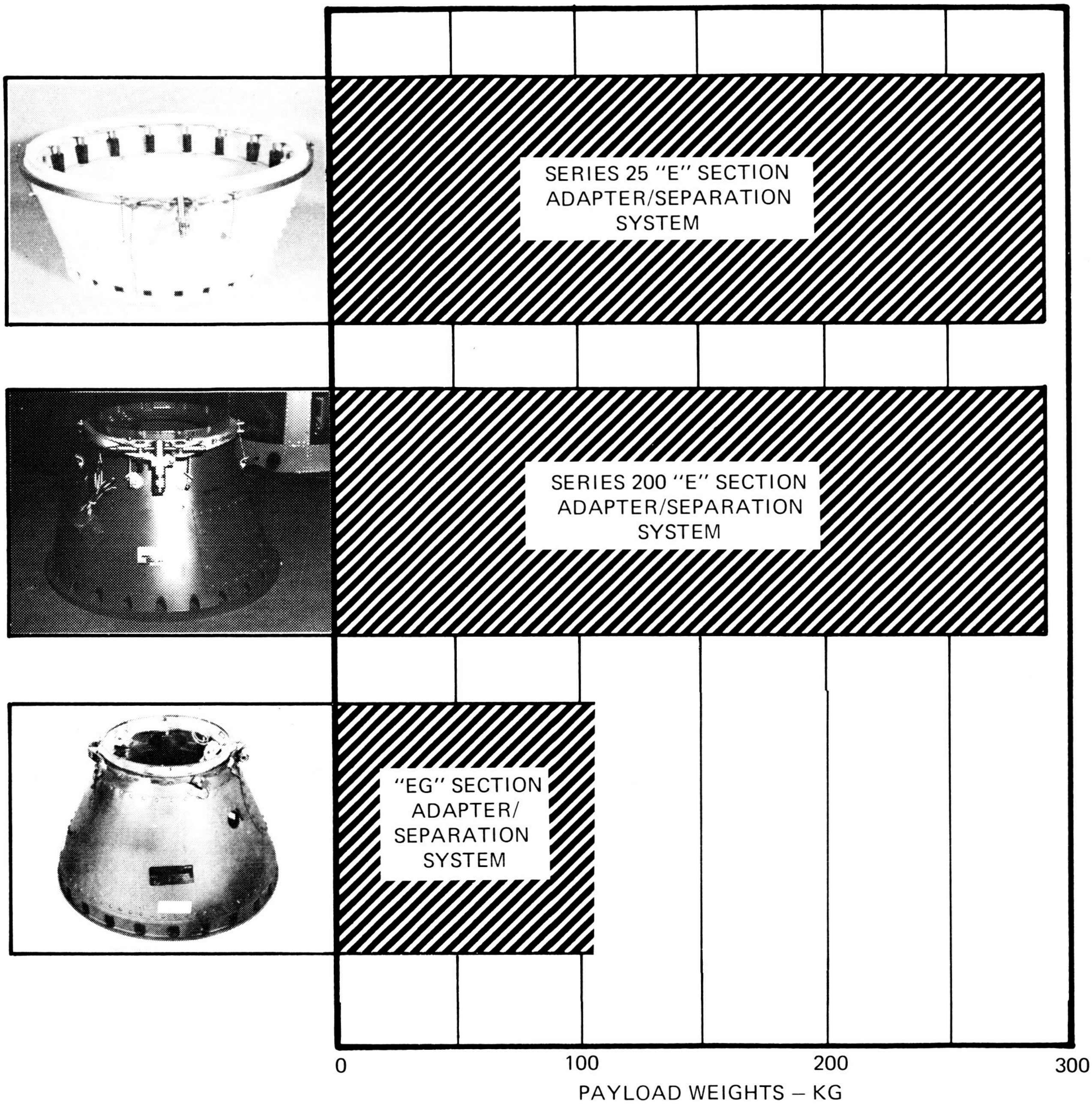
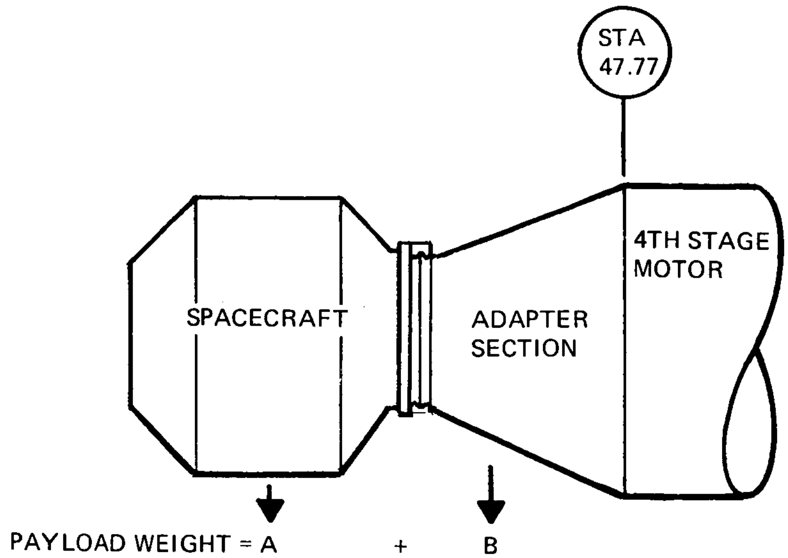


FIGURE V–2 ADAPTER CAPABILITY CHART

PAYLOAD TO VEHICLE ADAPTER SECTIONS



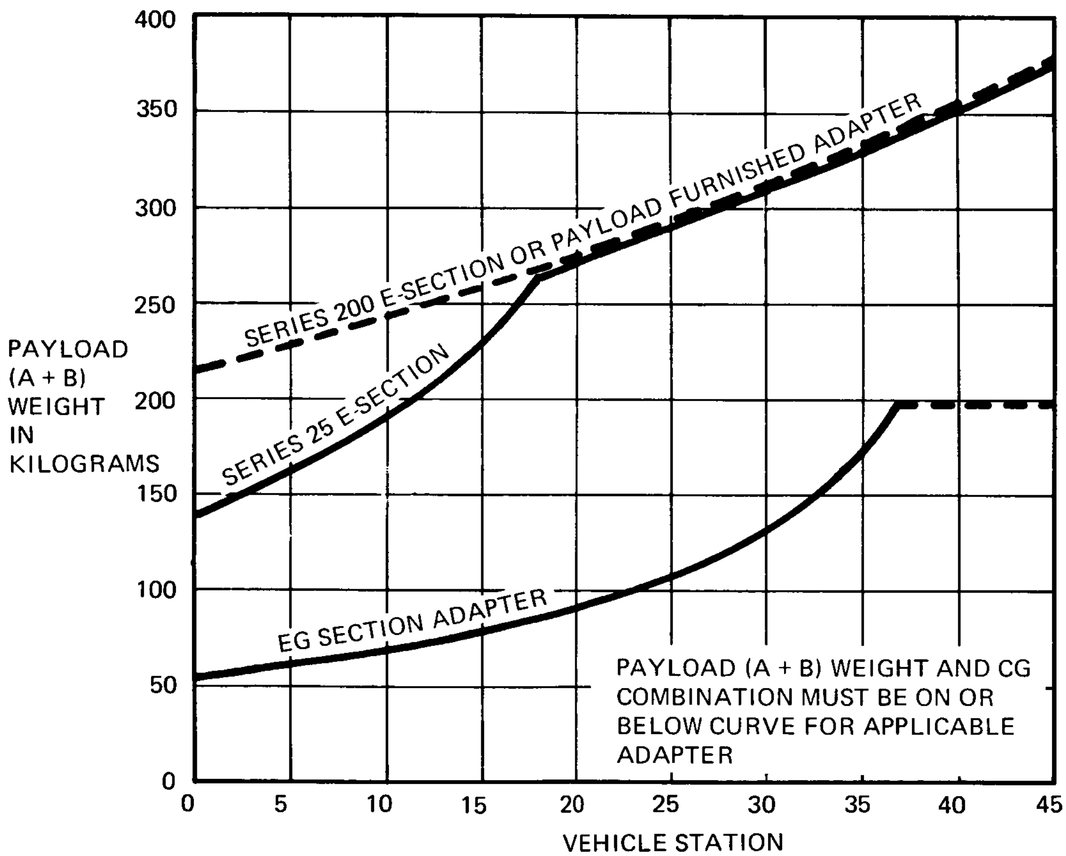


FIGURE V–3 ALLOWABLE PAYLOAD C.G. LOCATIONS

trajectory. Batteries and timers for the system are located on the Fourth Stage Module support ring on the aft flange of the fourth stage motor.

The separation system is remotely armed from the blockhouse through the umbilical. Separation timer initiation is switch actuated at fourth stage separation. The timers supply a signal to the dual, self-contained, pyrotechnic nuts on the payload separation clamp. Separation of the clamp releases the energy stored in the springs resulting in separation of the spacecraft from the adapter structure.

The standard timer is a four channel solid-state timer whose channels are independently adjustable in ten second increments from 10 to 9990 seconds within .05% accuracy. This timer is powered by a replaceable, self-contained battery which can be recharged from the blockhouse. The recommended time for payload separation is five minutes after fourth stage ignition.

Separation system batteries may be made available to power some payload functions (cable cutters, etc.) depending on power requirements.

An interface connector package consisting of brackets, electrical connectors, and harness is available for use at the option of the Payload Agency. This package allows the spacecraft to utilize available timer channels for performing such functions as despin, boom erection, spacecraft activation, etc., prior to separation from the fourth stage.

For ground operations, provisions are made to hold the separation springs in the restrained position to permit spacecraft installation and removal.

The ensuing paragraphs provide technical data for each adapter and further defines the sections and payload/adapter interfaces.

2.2 "EG" SECTION ADAPTER/ SEPARATION SYSTEM

The "EG" section adapter is a magnesium structure. The structure will support a payload weight of 100 kilograms at vehicle station 24. Figure V—4 shows the basic dimensions of the adapter and the separation clamp trajectory envelope.

The payload interface dimensional requirements for the payload support ring are shown in Figure V—5.

A weight summary of the "EG" section adapter/separation system is presented in Section 4 of this chapter.

Figure V—6 shows the separation velocity in meters-per-second versus spacecraft weight for the separation system. It should be noted that the spacecraft weight reflected in this curve includes only the separating mass; i.e., spacecraft and support ring.

Figure V-7 shows a typical "EG" section adapter/spacecraft installation.

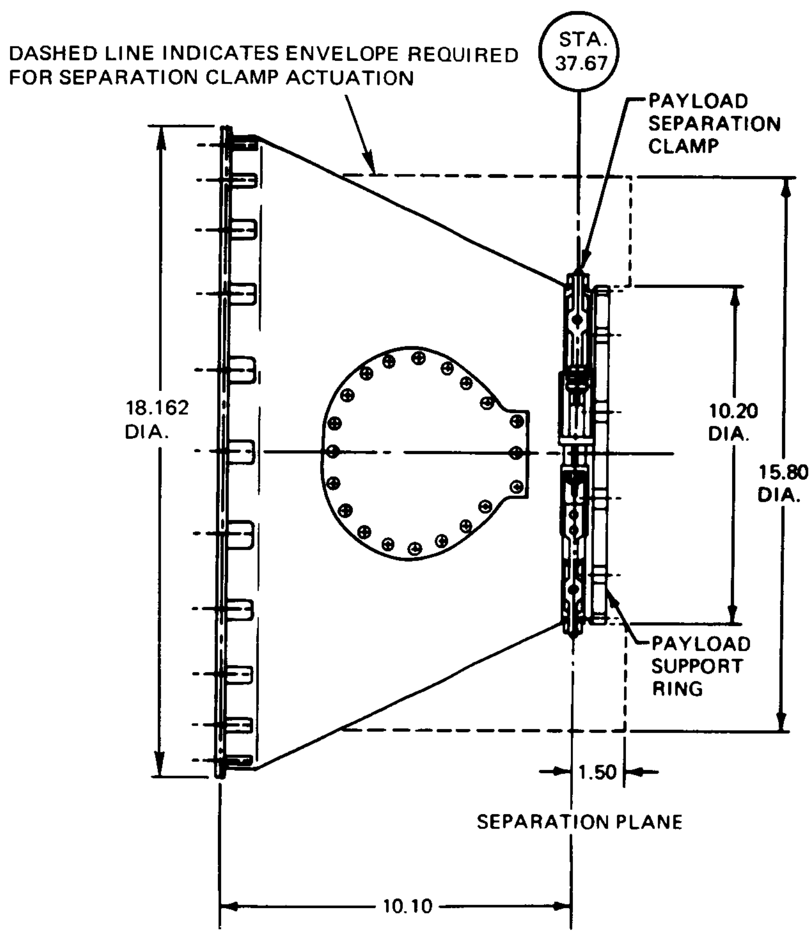
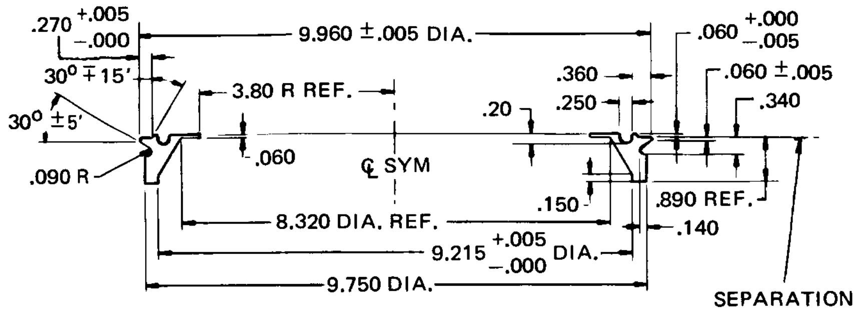


FIGURE V–4 "EG" SECTION ADAPTER



SECTION B-B

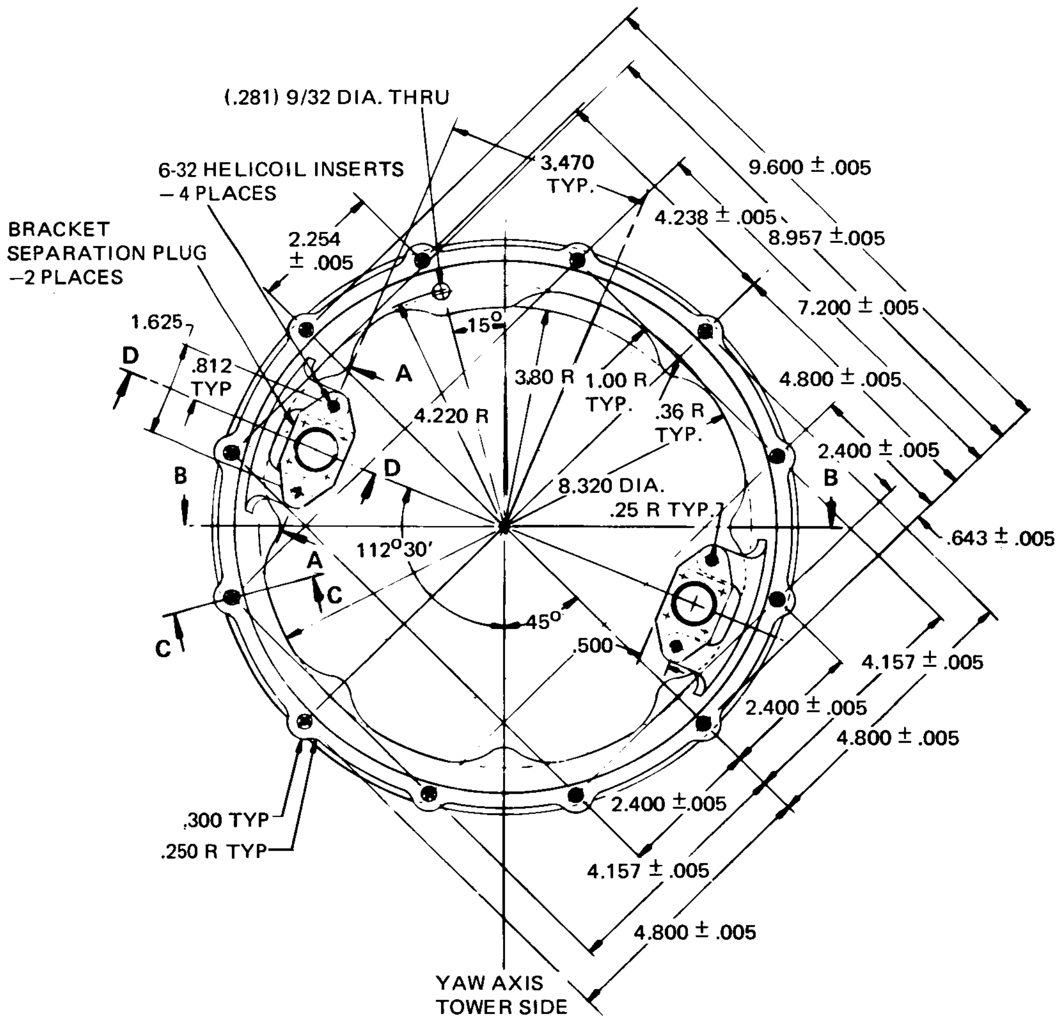
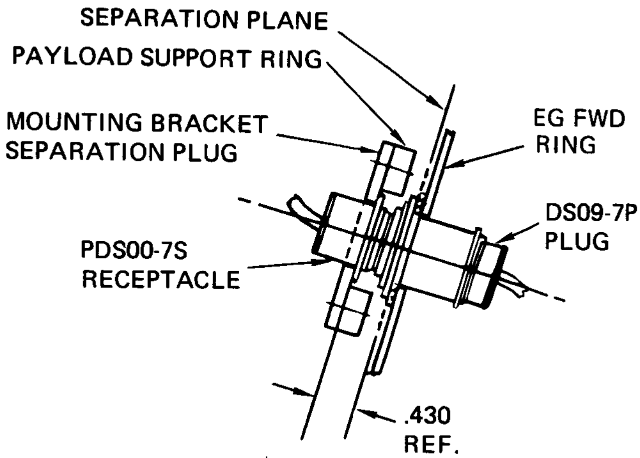
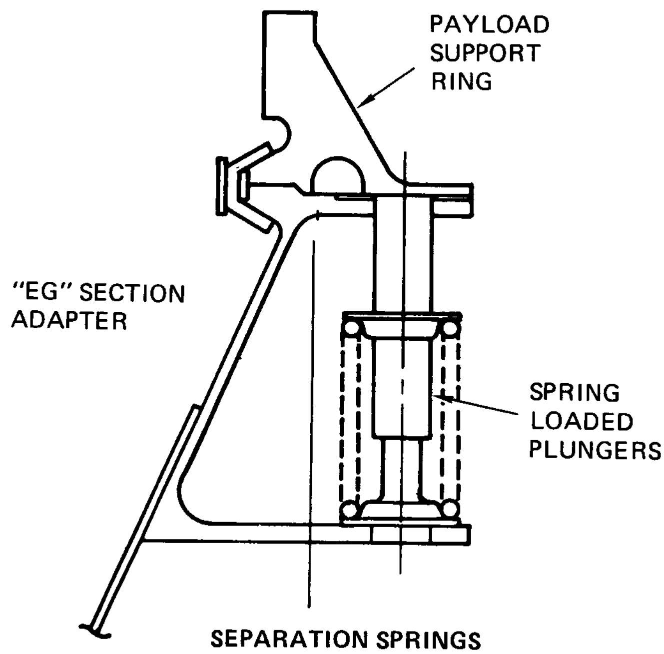


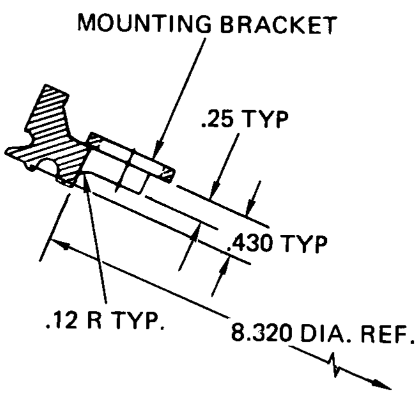
FIGURE V–5 "EG" SECTION ADAPTER INTERFACE (SHEET 1 OF 3)



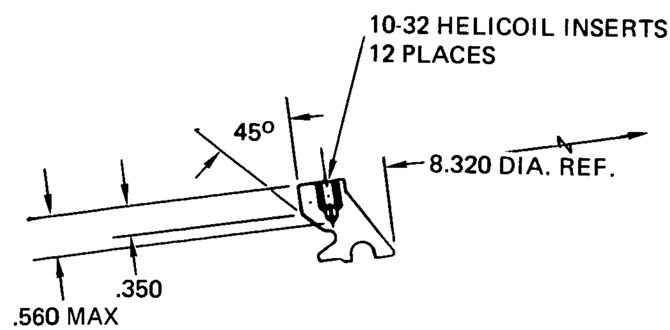
SECTION A-A



SEPARATION SPRINGS
TYPICAL 8 PLACES

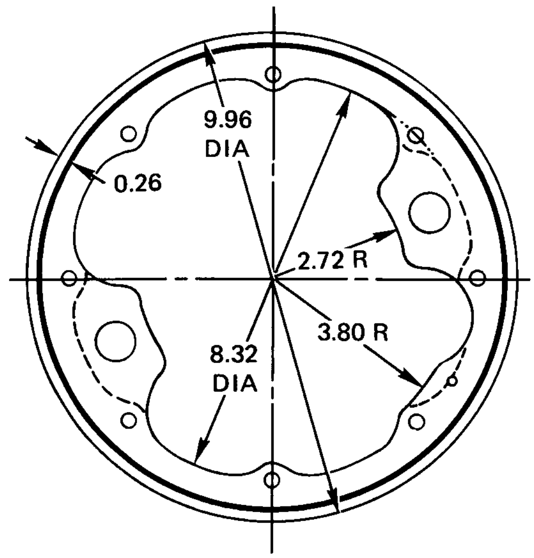


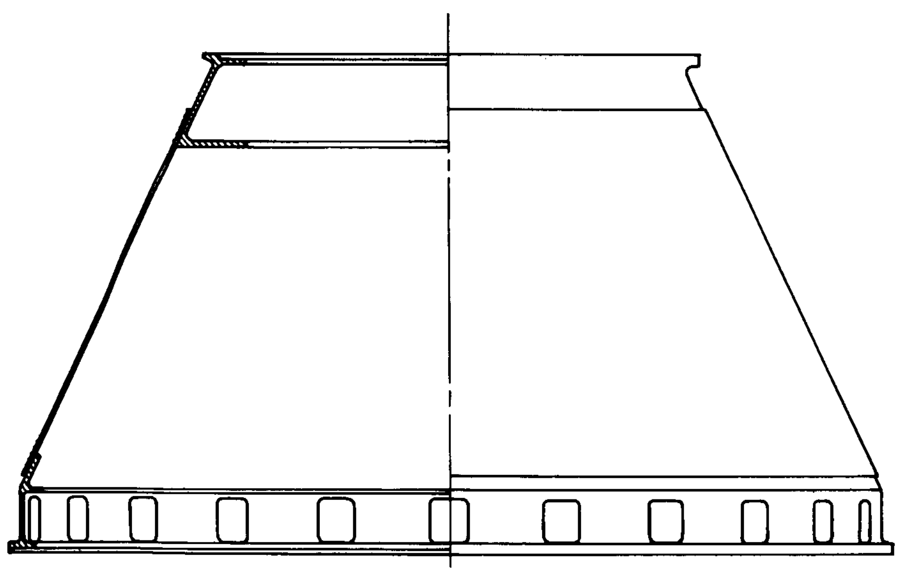
SECTION D-D
TYP. 2 PLACES 180° APART
PLUG REMOVED FOR CLARITY



SECTION C-C
TYP. 12 PLACES

FIGURE V–5 "EG" SECTION ADAPTER INTERFACE (SHEET 2 OF 3)





"EG" SECTION ADAPTER

FIGURE V–5 "EG" SECTION ADAPTER INTERFACE (SHEET 3 OF 3)

1. POTENTIAL ENERGY OF SPRINGS (8) = 0.912 KILOGRAM-METERS
2. SPRING EFFICIENCY = 77%
3. EXPENDED FOURTH STAGE WEIGHT:
   ALTAIR III + UPPER "D" SECTION + STANDARD "EG" SECTION + 4TH STAGE MODULE = 44.2 KILOGRAMS
4. SPACECRAFT WEIGHT IS TOTAL WEIGHT OF SEPARATING MASS (SPACECRAFT AND SUPPORT RING)

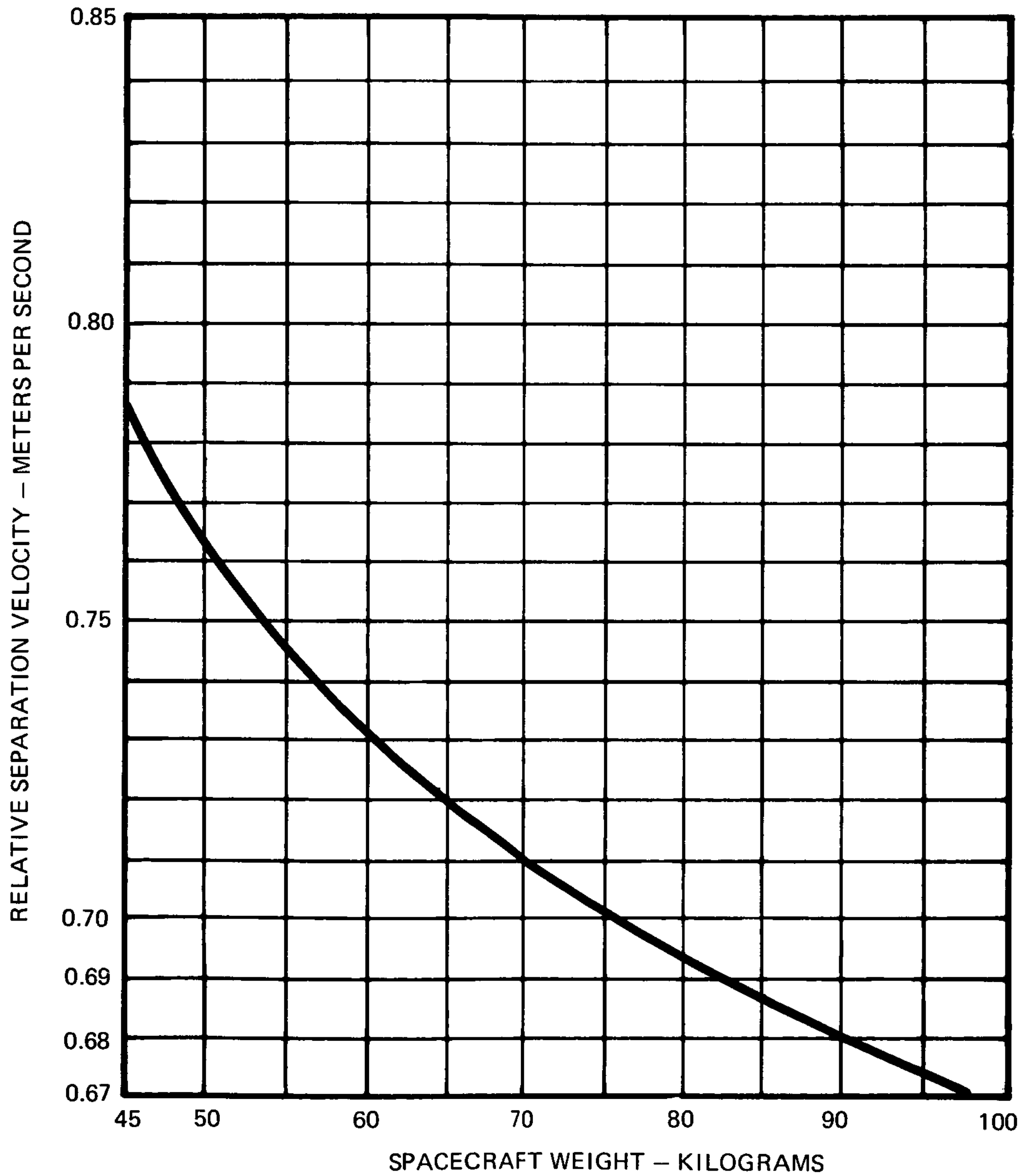


FIGURE V–6 RELATIVE SEPARATION VELOCITY OF SPACECRAFT AND EXPENDED FOURTH STAGE – "EG" SECTION

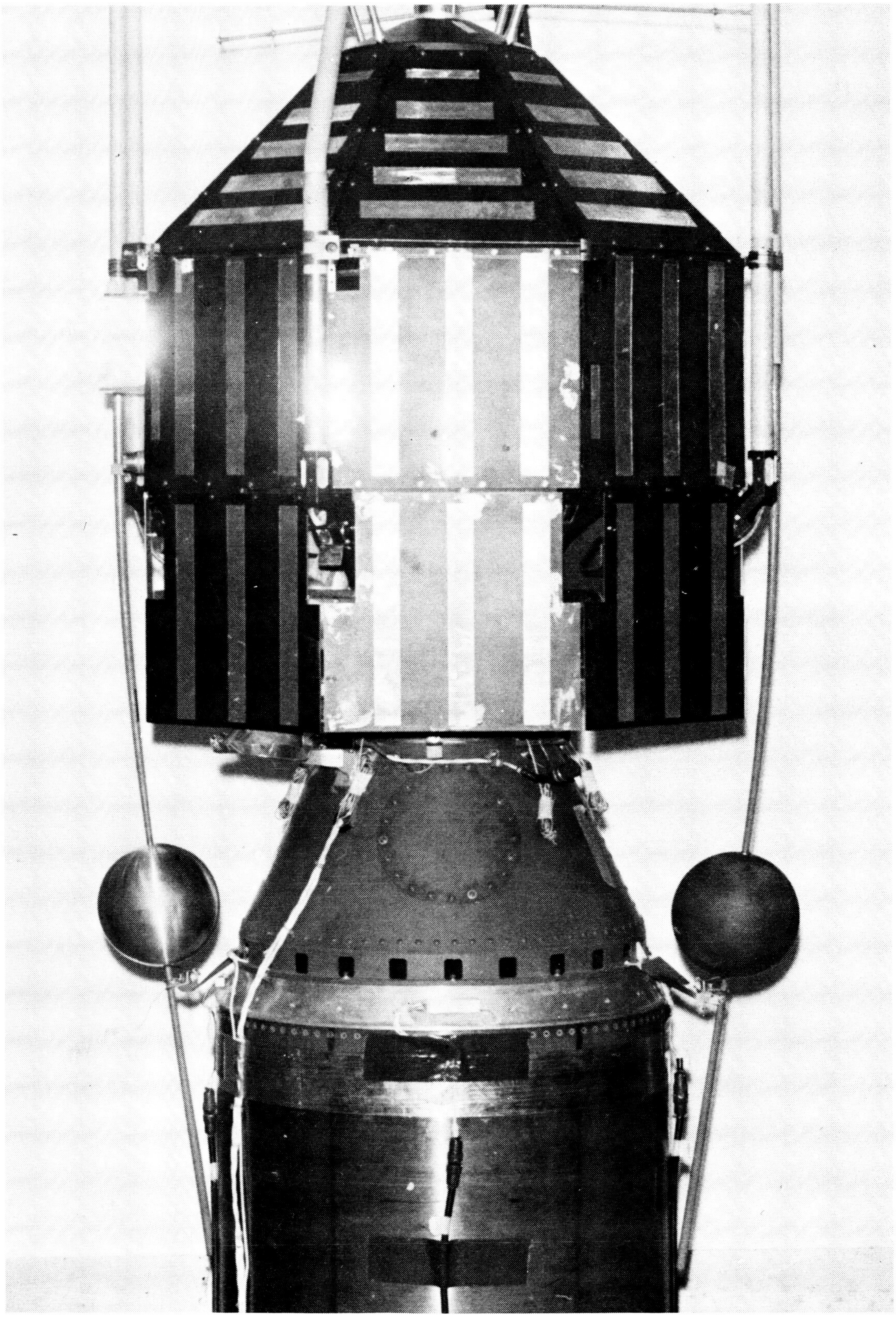


FIGURE V–7 "EG" SECTION ADAPTER/SPACECRAFT INSTALLATION

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2.3 SERIES 200 "E" SECTION ADAPTER/ SEPARATION SYSTEM

The Series 200 "E" section adapter is a magnesium structure. The structure will support a payload weight of 290 kilograms at vehicle station 24. Figure V–8 shows the basic dimensions of the adapter and the separation clamp trajectory envelope.

The payload interface dimensional requirements for the payload support ring are shown in Figure V–9.

A weight summary of the Series 200 "E" section adapter/separation system is presented in Section 4 of this chapter.

Figure V–10 shows the separation velocity in meters-per-second versus spacecraft weight for the separation system. It should be noted that the spacecraft weight reflected in this curve includes only the separating mass; i.e., spacecraft and support ring.

Figure V–11 shows a typical Series 200 "E" section adapter/spacecraft installation.

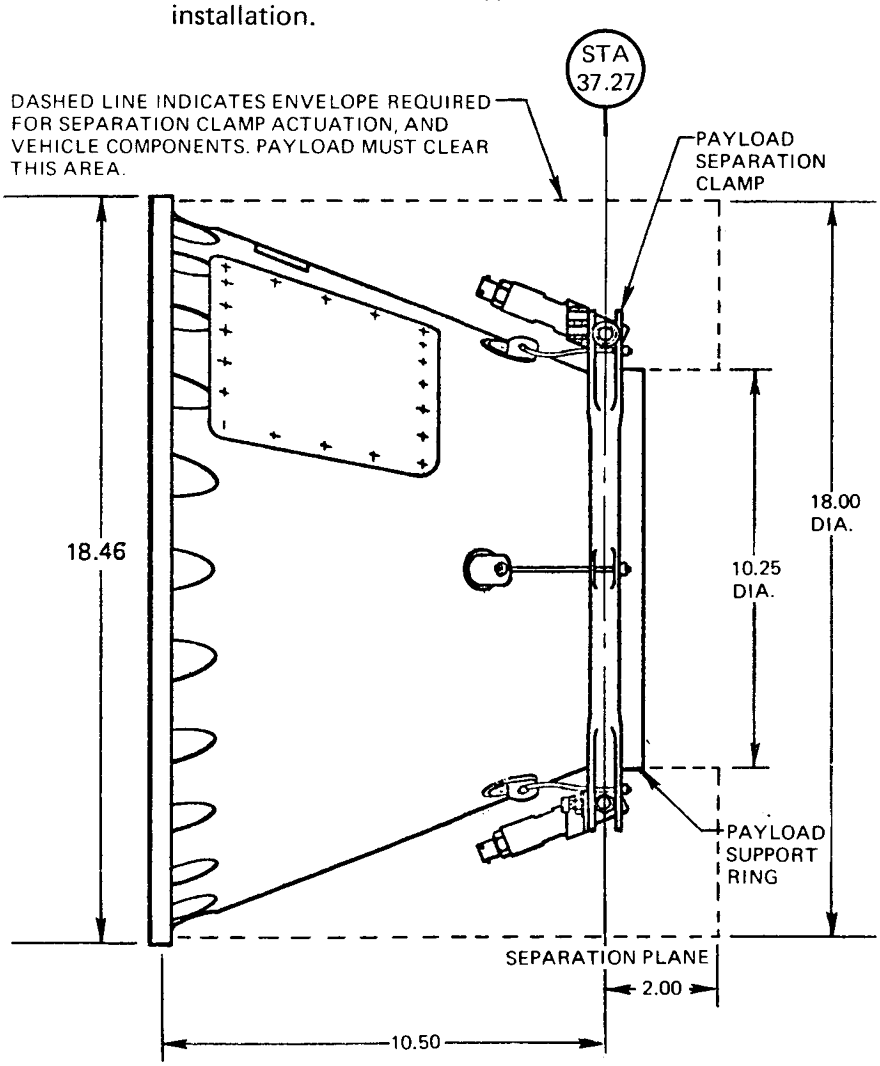
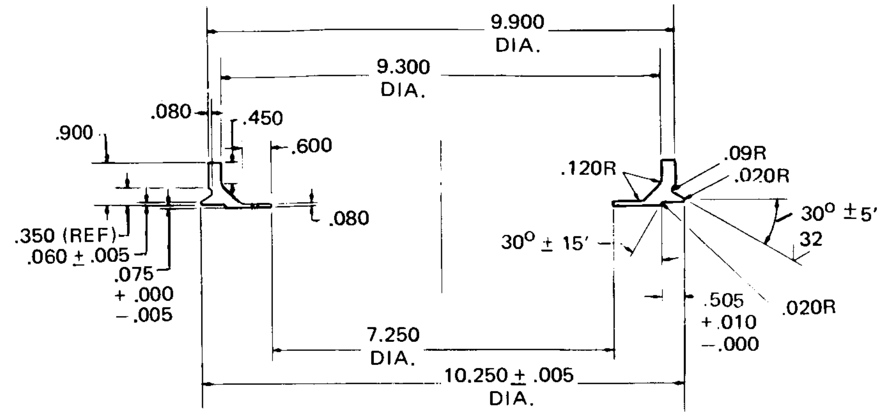


FIGURE V–8 SERIES 200 "E" SECTION ADAPTER



SECTION A–A

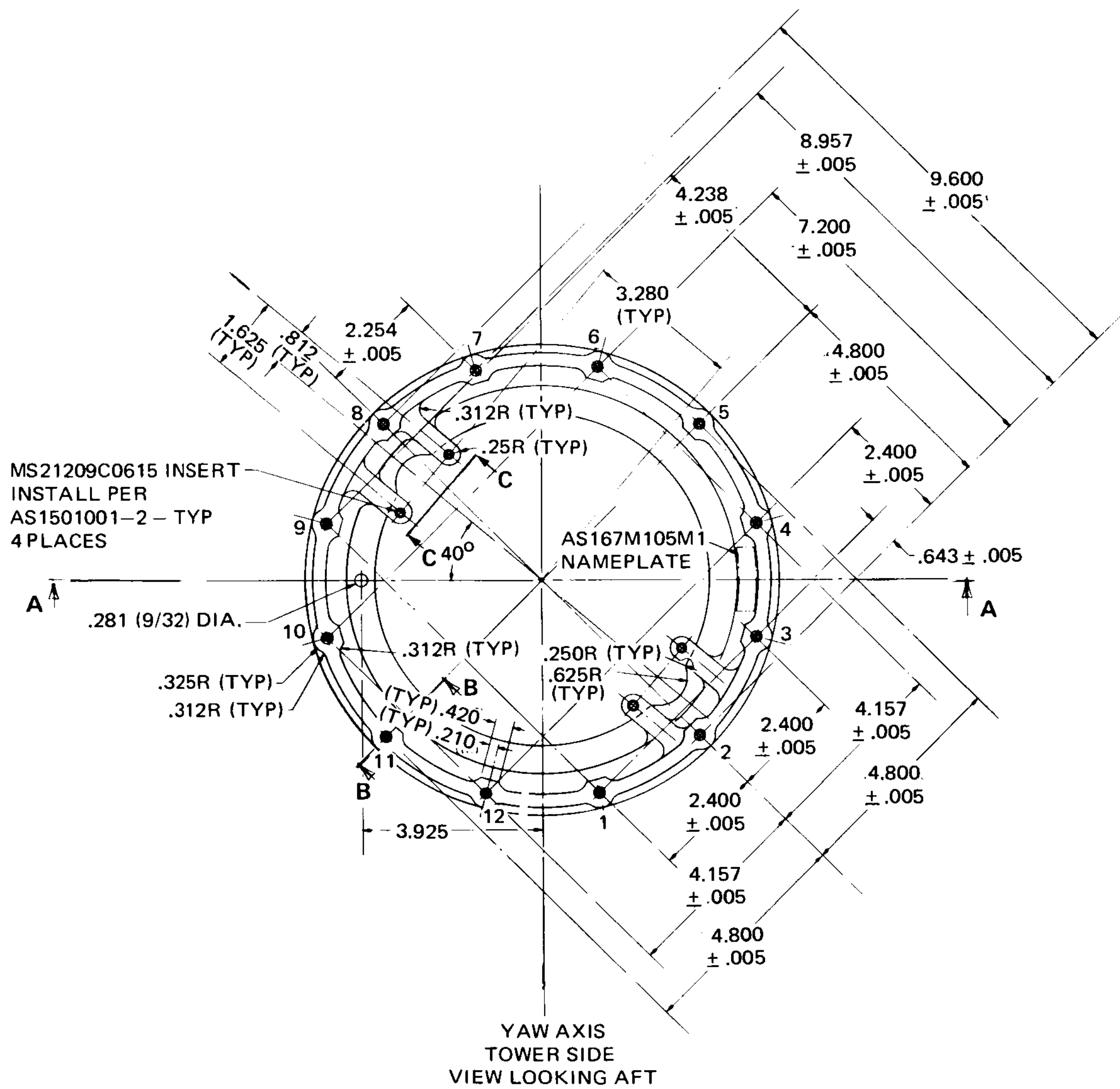
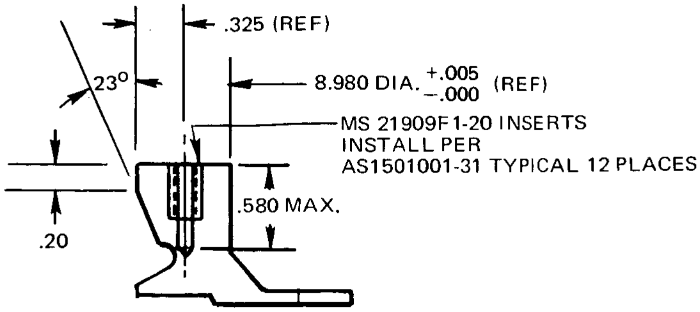
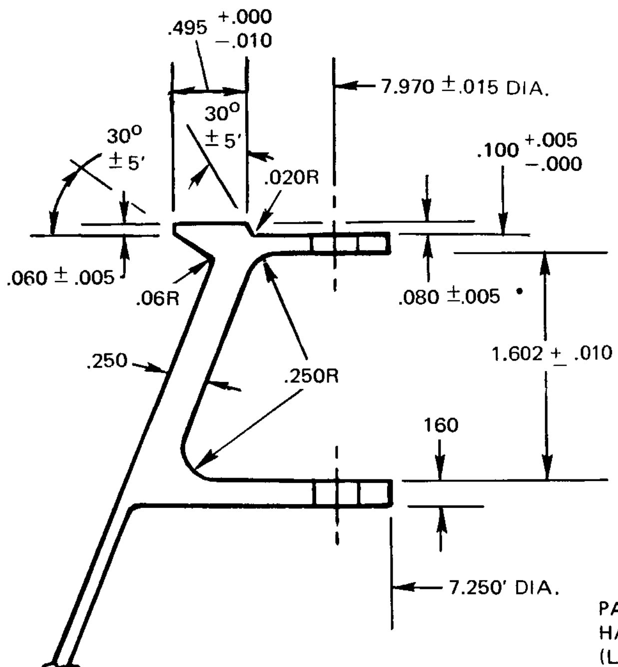


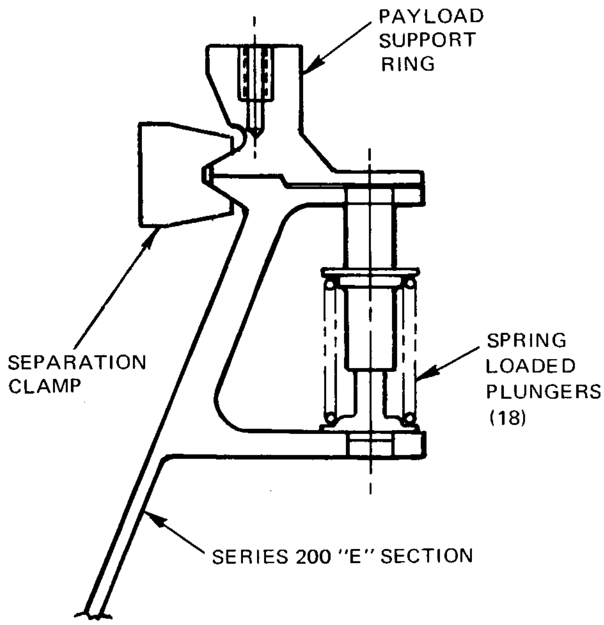
FIGURE V–9 SERIES 200 "E" SECTION ADAPTER INTERFACE (SHEET 1 OF 3)



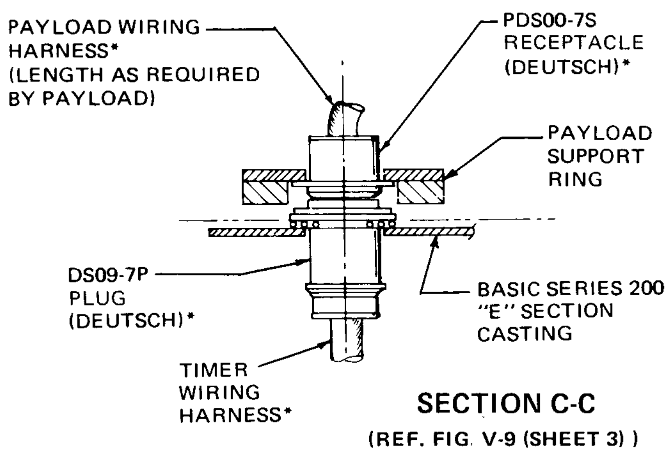
SECTION B-B
(REF. FIG. V-9 (SHEET 1))



DETAIL A
(REF. FIG. V-9 (SHEET 3))

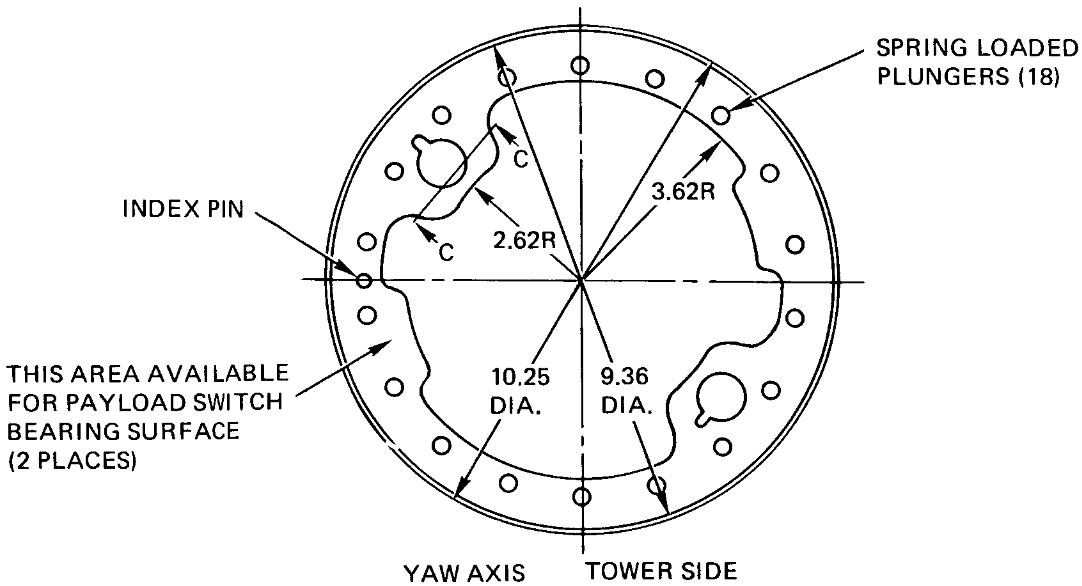


SEPARATION SPRINGS
(18)

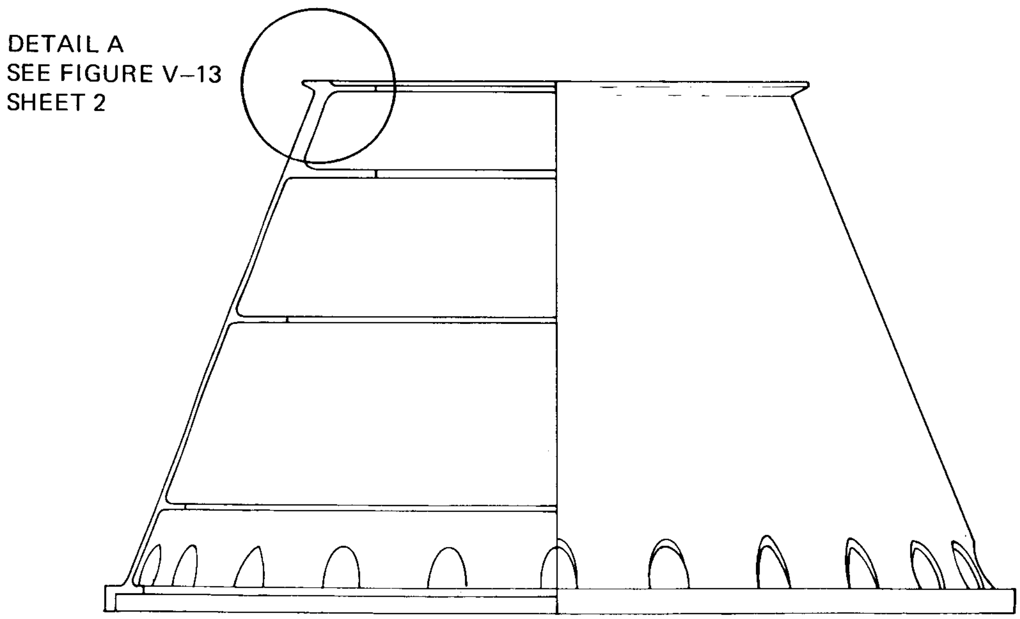


SECTION C-C
(REF. FIG. V-9 (SHEET 3))
TYPICAL 2 PLACES, IF REQUIRED

FIGURE V–9 SERIES 200 "E" SECTION ADAPTER INTERFACE (SHEET 2 OF 3)



PAYLOAD ADAPTER
ATTACHING SURFACE
LOOKING AFT



BASIC ADAPTER

FIGURE V–9 SERIES 200 "E" SECTION ADAPTER INTERFACE (SHEET 3 OF 3)

NOTE:

1. POTENTIAL ENERGY OF SPRINGS (18) = 1.94 KILOGRAM-METERS
2. SPRING EFFICIENCY = 76%
3. EXPENDED FOURTH STAGE WEIGHT:
   ALTAIR III + UPPER "D" SECTION + SERIES 200 "E" SECTION + 4TH STAGE MODULE = 47.0 KILOGRAMS
4. SPACECRAFT WEIGHT IS TOTAL WEIGHT OF SEPARATING MASS (SPACECRAFT AND SUPPORT RING)

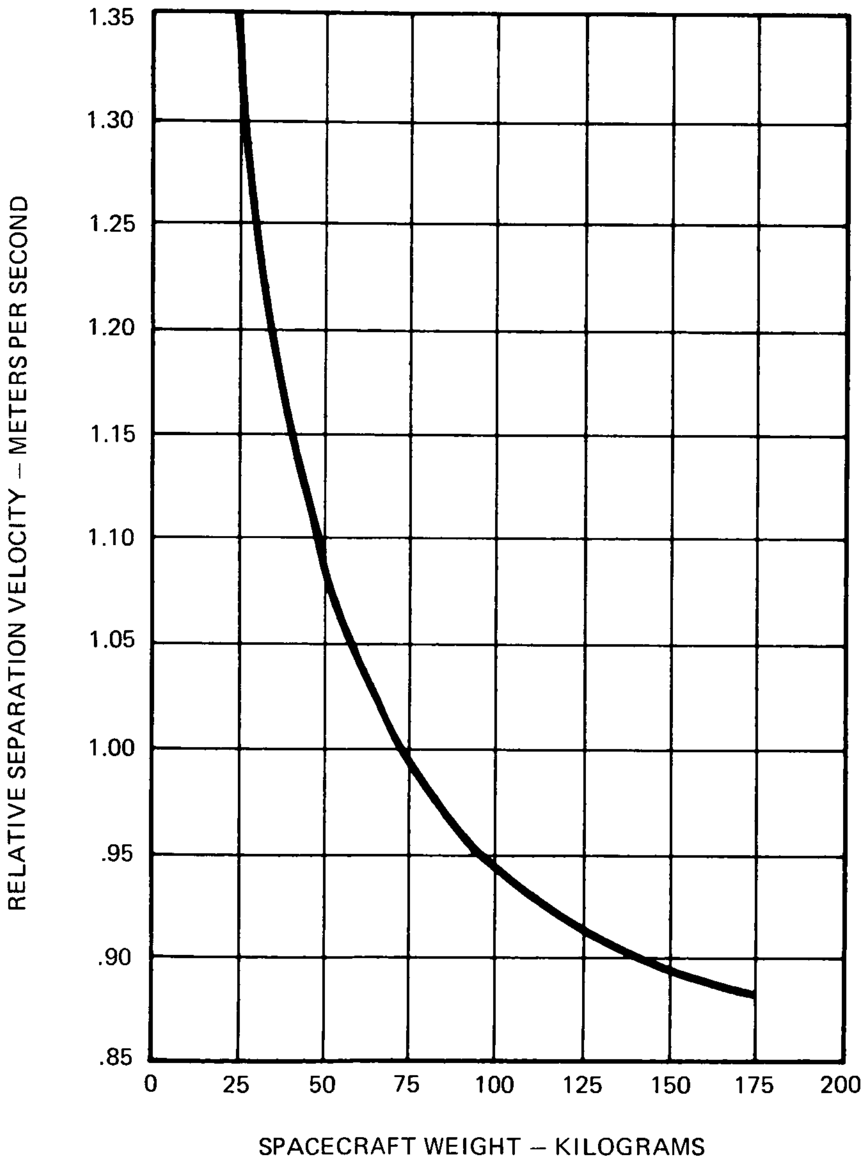


FIGURE V–10 RELATIVE SEPARATION VELOCITY OF SPACECRAFT AND EXPENDED FOURTH STAGE – SERIES 200 "E" SECTION

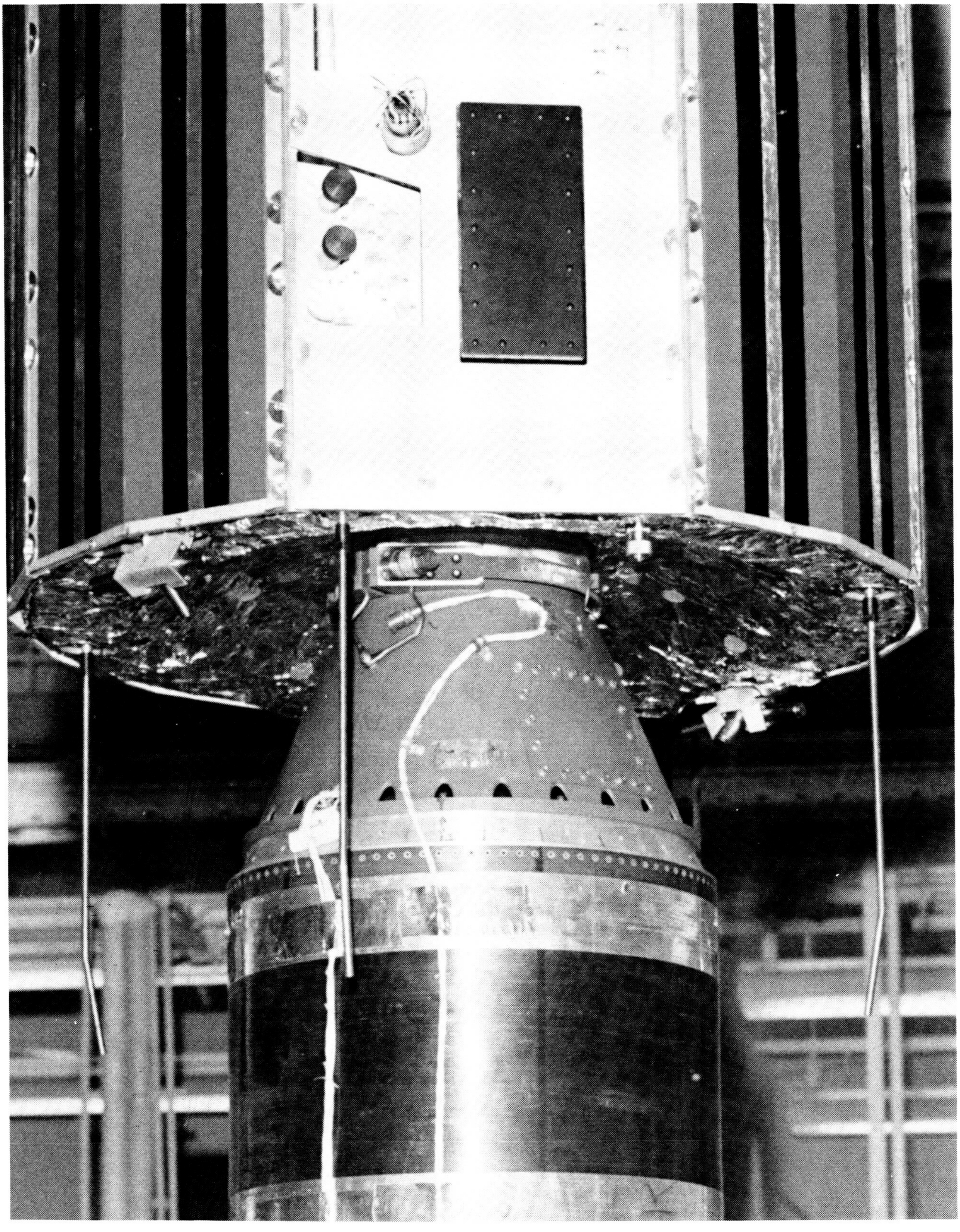


FIGURE V–11 SERIES 200 "E" SECTION ADAPTER/SPACECRAFT INSTALLATION

PAYLOAD TO VEHICLE ADAPTER SECTIONS

2.4 SERIES 25 "E" SECTION ADAPTER/ SEPARATION SYSTEM

The Series 25 "E" section adapter is a magnesium structure. The structure will support a payload weight of 290 kilograms at vehicle station 24. Figure V–12 shows the basic dimensions of the adapter and the separation clamp trajectory envelope.

The payload interface dimensional requirements for the payload support ring are shown in Figure V–13.

A weight summary of the Series 25 "E" section adapter/separation system is presented in Section 4 of this chapter.

Figure V–14 shows the separation velocity in meters-per-second versus spacecraft weight for the separation system. It should be noted that the spacecraft weight reflected in this curve includes only the separating mass; i.e., spacecraft and support ring.

Figure V–15 shows a typical Series 25 "E" section adapter/spacecraft installation.

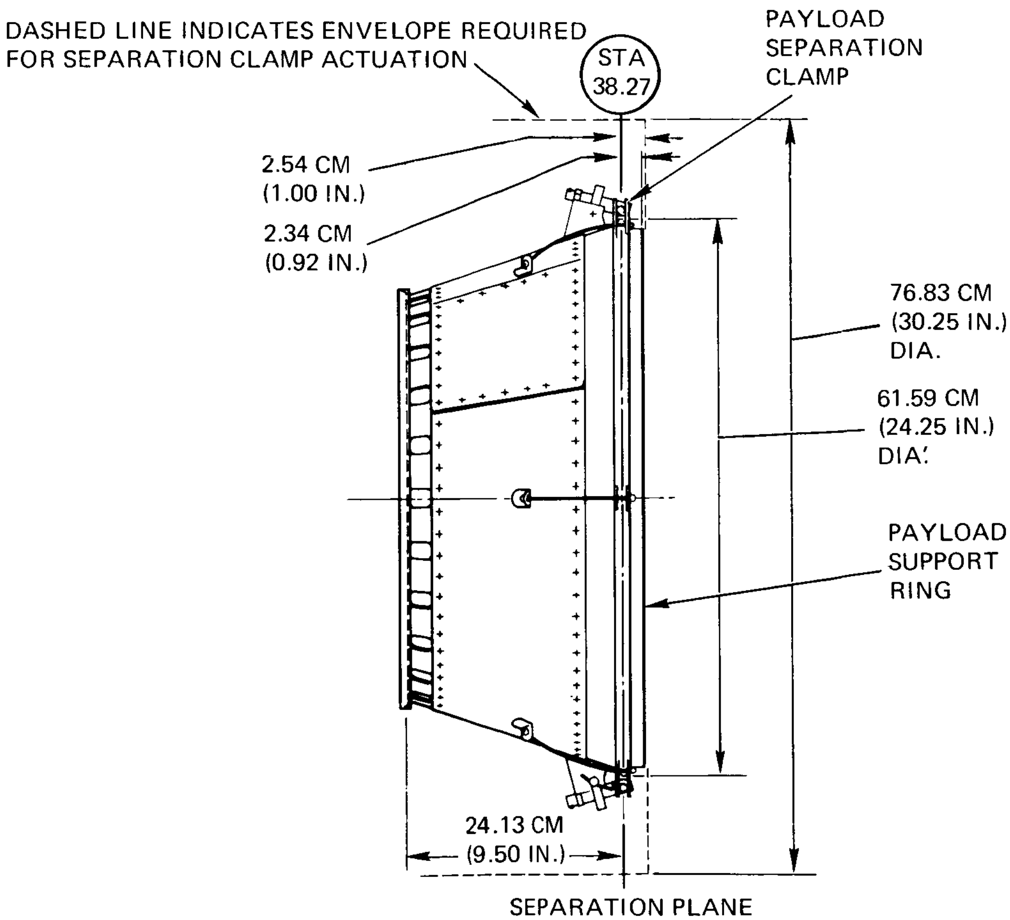


FIGURE V–12 SERIES 25 "E" SECTION ADAPTER

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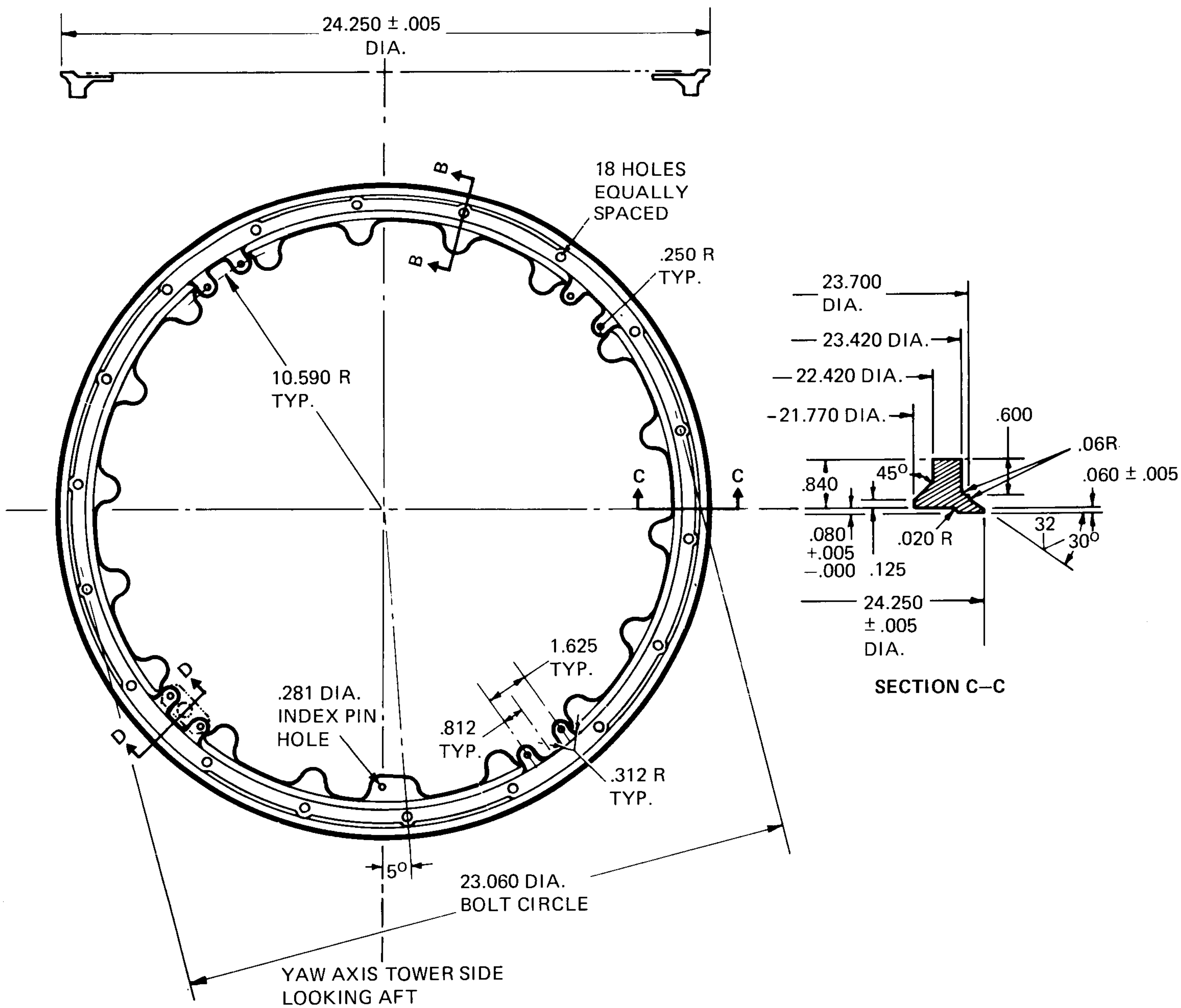

FIGURE V–13 SERIES 25 "E" SECTION ADAPTER INTERFACE (SHEET 1 OF 3)

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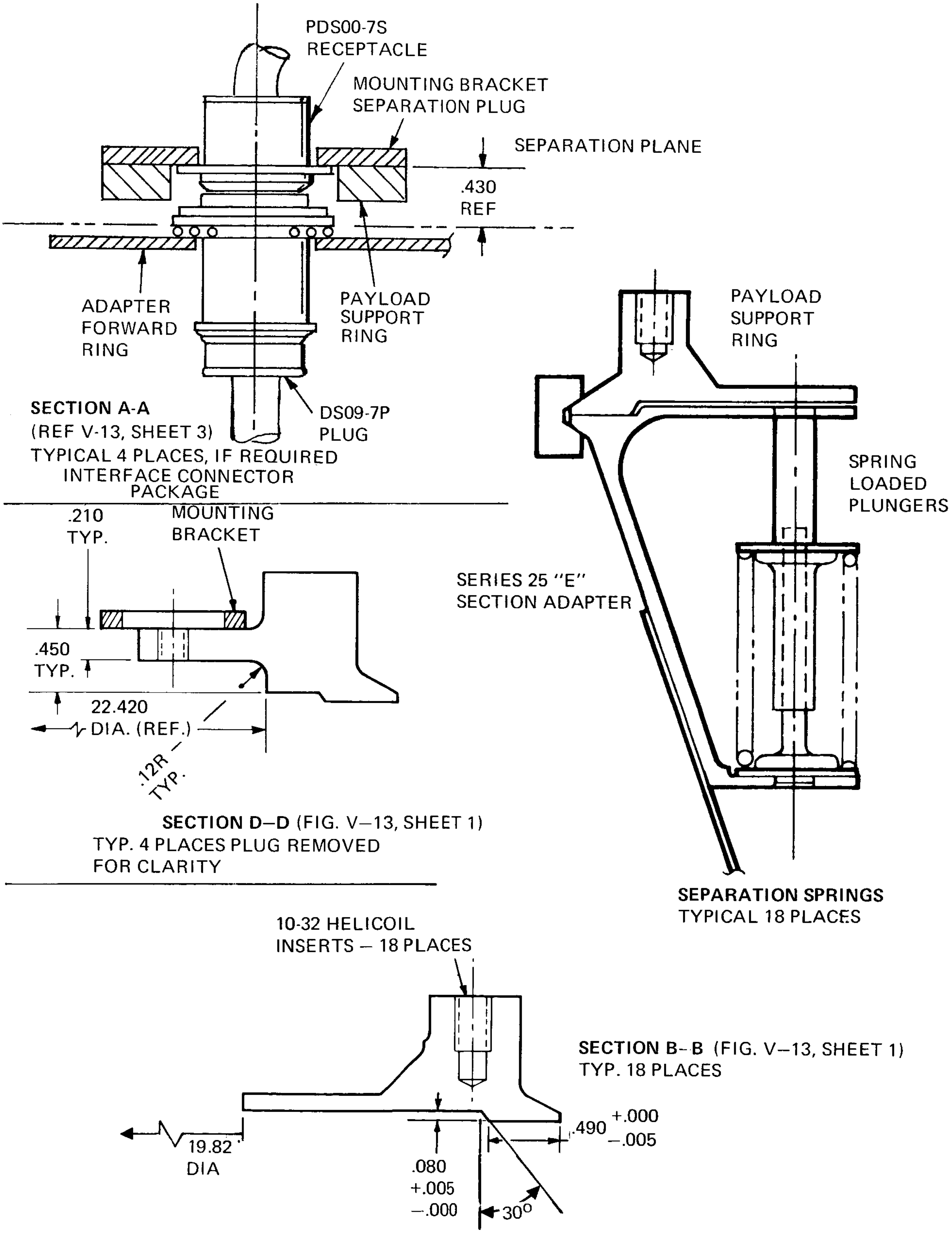

FIGURE V–13 SERIES 25 "E" SECTION ADAPTER INTERFACE (SHEET 2 OF 3)

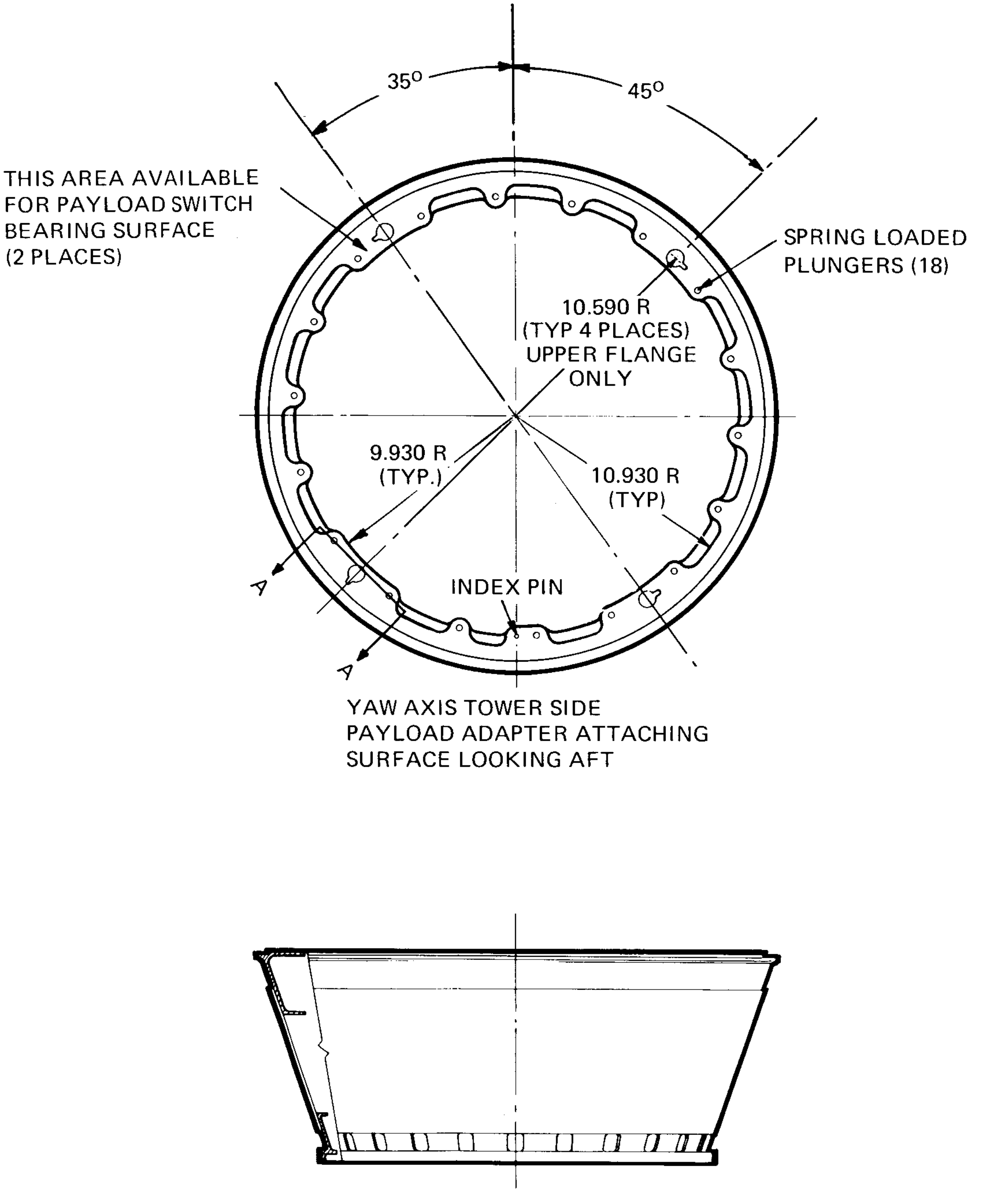

FIGURE V–13 SERIES 25 "E" SECTION ADAPTER INTERFACE (SHEET 3 OF 3)

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1. POTENTIAL ENERGY OF SPRINGS (18) = 6.81 KILOGRAM-METERS
2. SPRING EFFICIENCY = 75%
3. EXPENDED FOURTH STAGE WEIGHT:
   ALTAIR III + UPPER "D" SECTION + SERIES 25 "E" SECTION + 4TH STAGE MODULE = 47.9 KILOGRAMS
4. SPACECRAFT WEIGHT IS TOTAL WEIGHT OF SEPARATING MASS (SPACECRAFT AND SUPPORT RING)

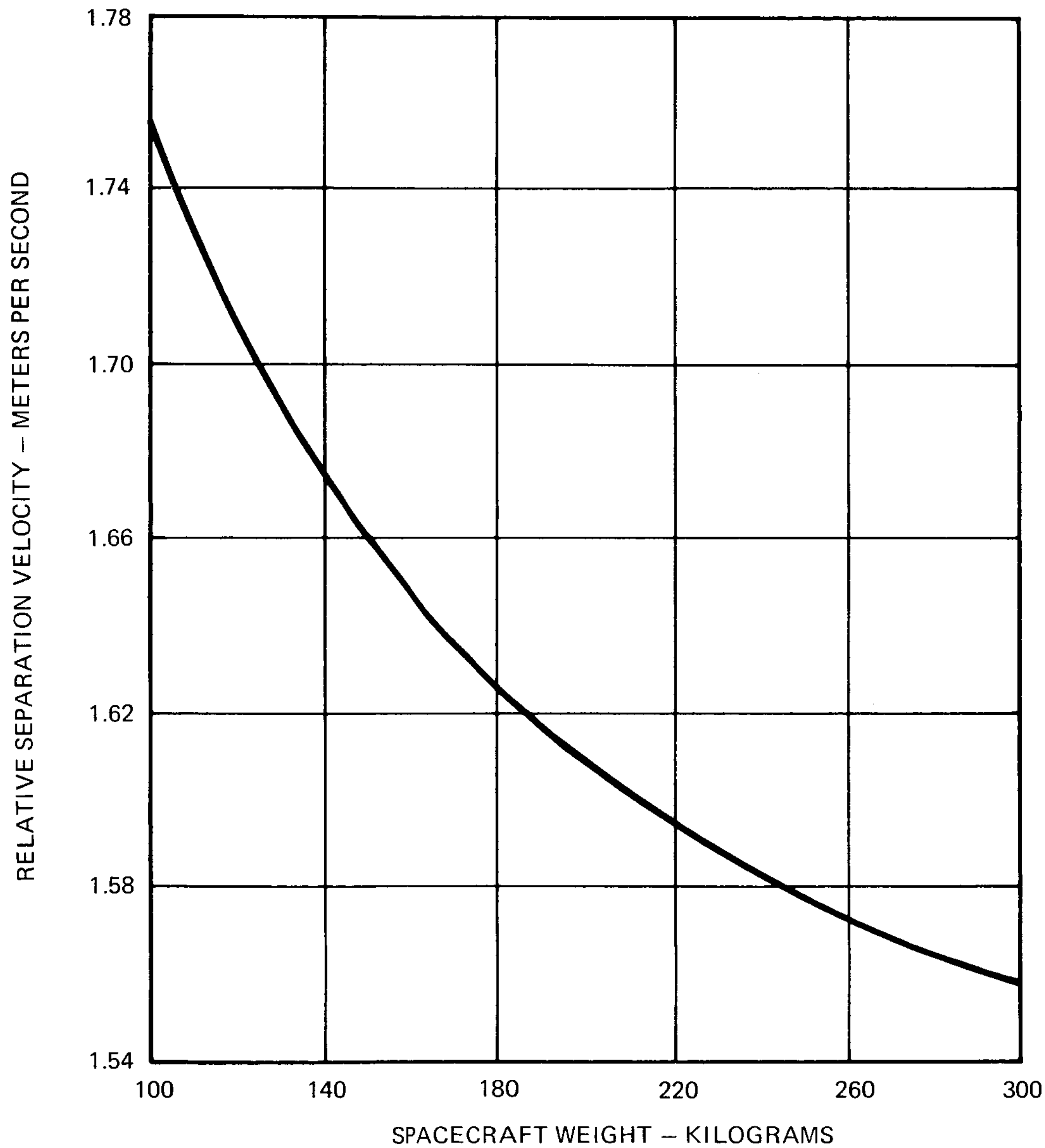

FIGURE V–14 RELATIVE SEPARATION VELOCITY OF SPACECRAFT AND EXPENDED FOURTH STAGE – SERIES 25 "E" SECTION

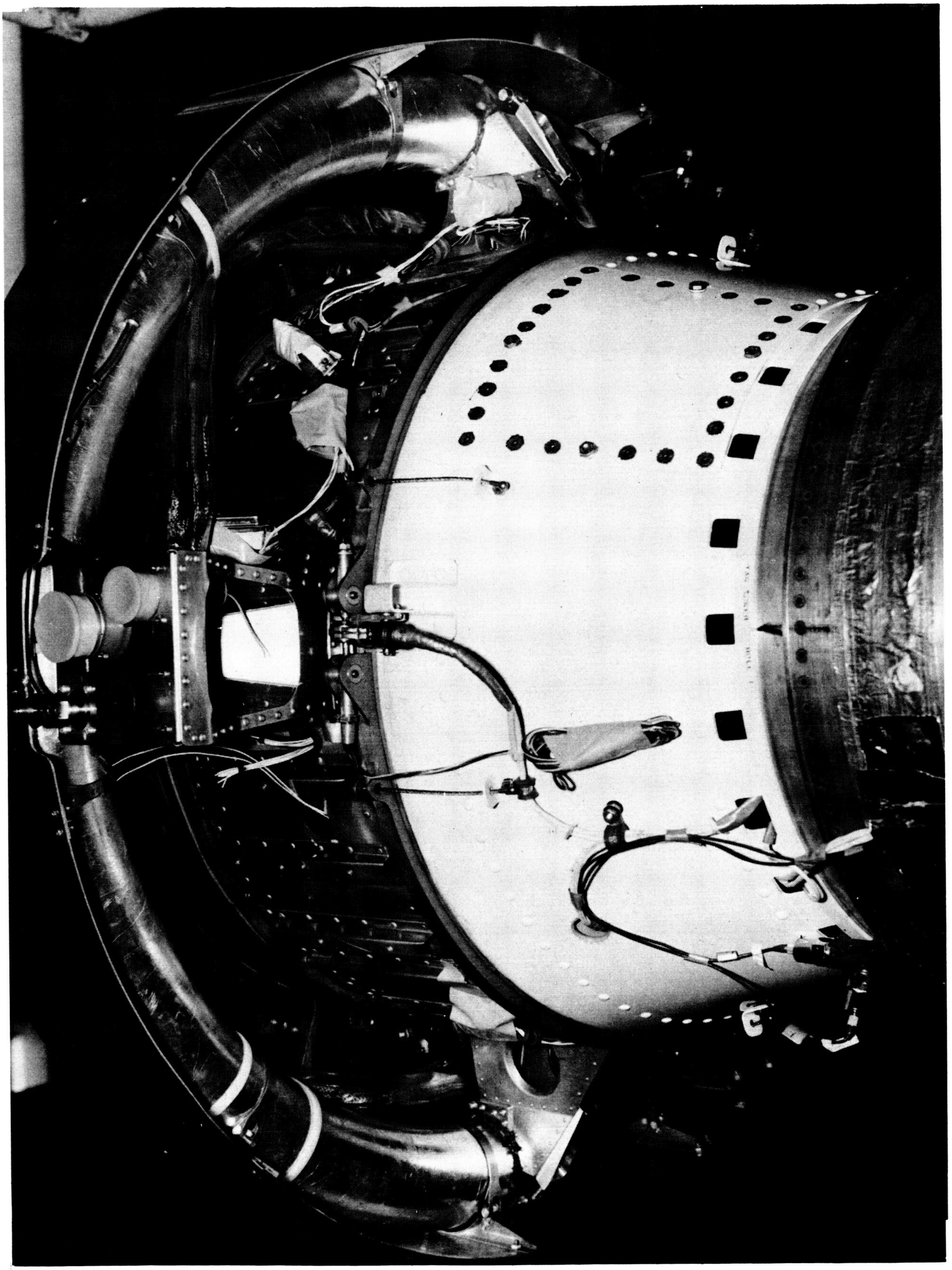

FIGURE V-15 SERIES 25 "E" SECTION ADAPTER/SPACECRAFT INSTALLATION

2.5 PAYLOAD AGENCY FURNISHED ADAPTER/ SEPARATION SYSTEM

The Payload Agency may choose to provide its own adapter/separation system in preference to one of the available basic systems. A Payload Agency furnished adapter/separation system must meet the design criteria specified herein, and must be closely coordinated with Vought, through the Mission Working Group. The design of a payload Agency furnished separation system must be approved in writing by the Mission Working Group prior to its use with the Scout Vehicle.

The payload may be attached directly to the final stage motor forward flange with its own mating adapter. The interface dimensional requirements for the fourth stage motor forward flange are shown in Figure V—16. The adapter must contain provisions for access to the final stage motor head-end ignition squibs. The standard adapters shown may be used as a guide for determining the general size and location of access openings. Two small openings (approximately 0.75 inch diameter holes) will be required for routing of the head-end ignition squib harness through the adapter for attachment to the final stage motor. Grommets are required in the holes for protection of the harnesses.

The timer actuating circuit with switch located on the "D" section separation plane may be used by a Payload Agency furnished adapter/ separation system (see Figure V—17).

The maximum payload weight in relation to center-of-gravity location for Payload Agency furnished adapter/separation system is shown in Figure V-3.

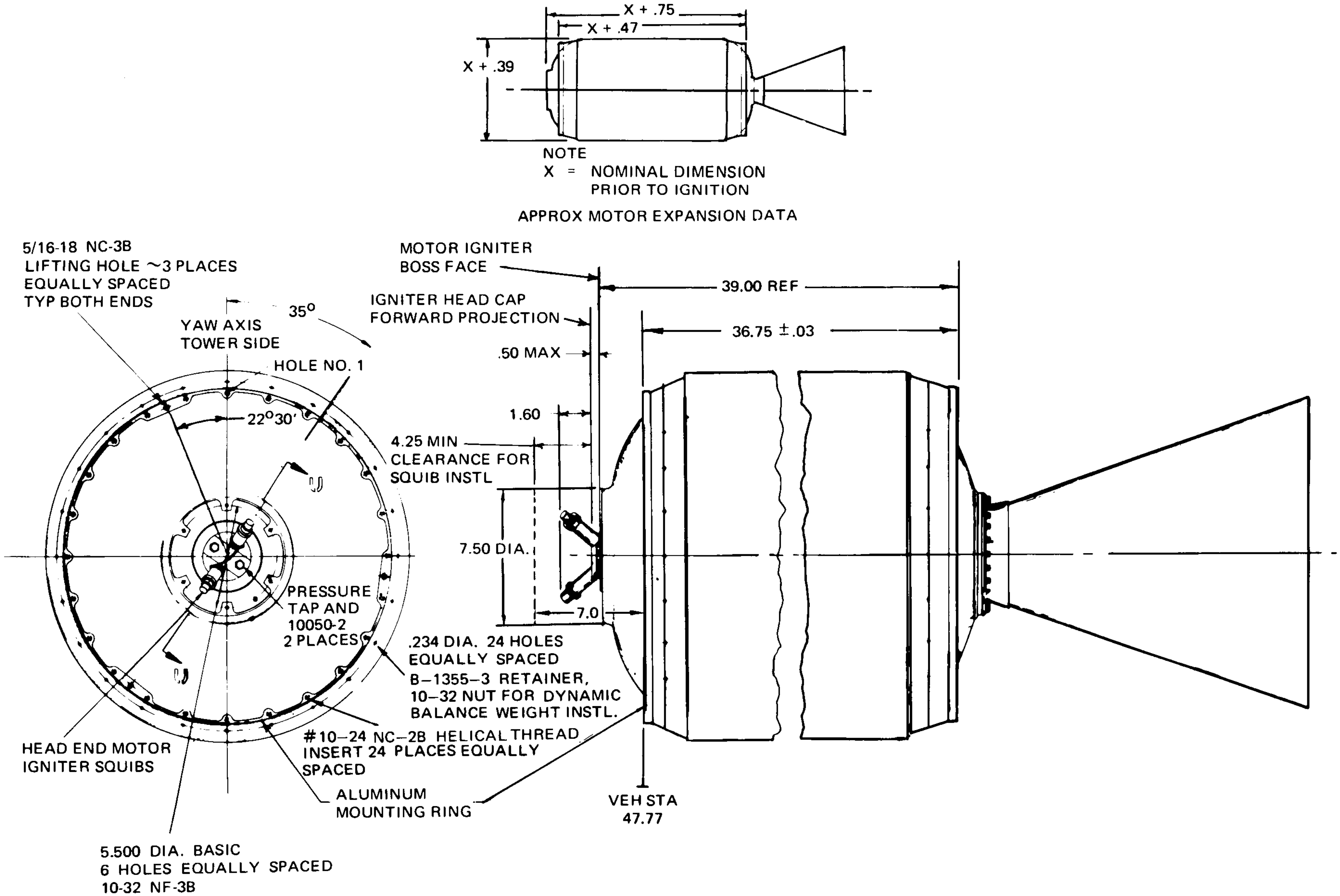

FIGURE V–16 SCOUT FOURTH STAGE MOTOR – ALTAIR III (SHEET 1 OF 2)

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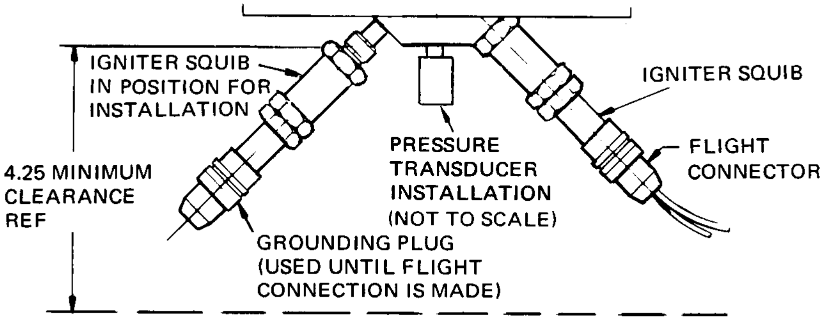
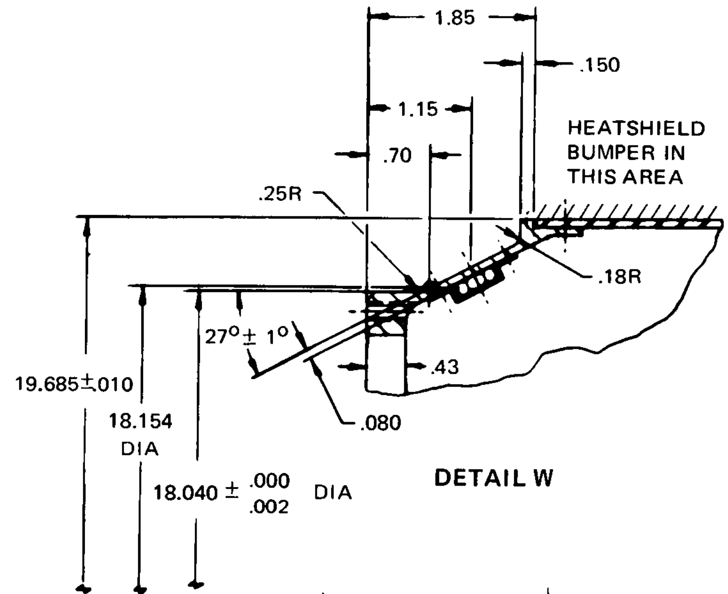


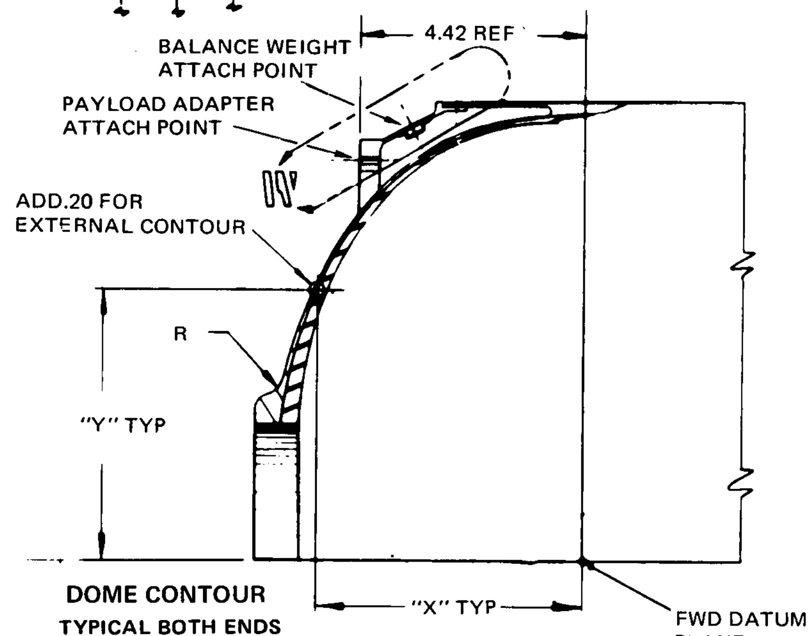
DETAIL U-U



DETAIL W

INTERNAL COORDINATES
TYP EACH END

X	Y
0.0	9.720
.181	9.715
.361	9.704
.631	9.676
.990	9.616
1.347	9.530
1.700	9.418
2.049	9.280
2.393	9.115
2.732	8.922
3.064	8.701
3.389	8.450
3.706	8.162
4.013	7.852
4.311	7.499
4.599	7.105
4.876	6.664
5.139	6.164
5.400	5.585
5.827	4.365



DOME CONTOUR
TYPICAL BOTH ENDS

FIGURE V–16 SCOUT FOURTH STAGE MOTOR – ALTAIR III (SHEET 2 OF 2)

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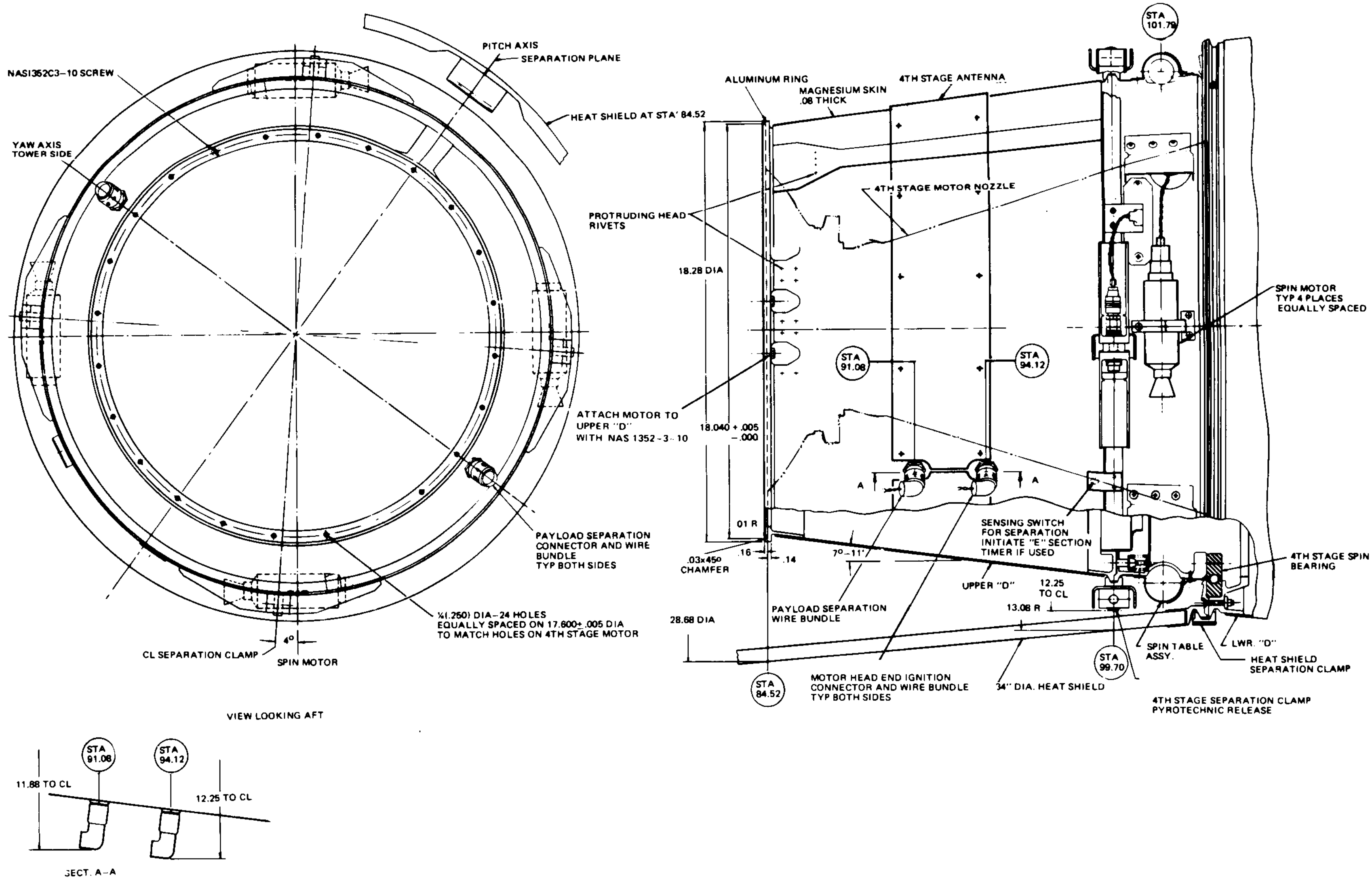

FIGURE V–17 SCOUT UPPER "D" TRANSITION SECTION

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3.0 HEATSHIELDS

Presented in this section are the heatshields which are available for Scout vehicles. The heatshield provides environmental protection to the payload in addition to carrying fourth stage structural loads during atmospheric flight. A coast period of five seconds minimum follows second stage burnout during which the heatshield is ejected.

Three heatshields are available for use with the Scout vehicles and are identified as follows:

34 inch diameter, minus 25 inch nose station (.86 meter dia.)

34 inch diameter, minus 40 inch nose station (.86 meter dia.)

42 inch diameter, minus 45 inch nose station (1.07 meters dia.)

3.1 DESCRIPTION AND OPERATION

The heatshields are a fiberglass laminate and honeycomb composite split shell structure, with a metal nose cap attached to the launch tower side half shell. Steel half rings at the base of the shells accept a joining clamp. A typical heatshield assembly is shown in Figure V–18.

A bumper, located inside the heatshield at the forward flange of the fourth stage motor case, distributes loads between the heatshield and the motor. Cutouts in the bumper for payload envelope require detailed coordination because of the important structural role of the bumper itself. The bumper may be located at a different vehicle station than shown if the Payload Agency will provide a mating surface on the payload. This subject will be discussed in detail at Mission Working Group meetings.

The exterior of the heatshield, forward of the intersection of the cylindrical and aft conical sections, is covered with cork. The exterior is then painted with Government Specification TT-L-32 Insignia White Lacquer.

The interior of the heatshield is painted with Military Specification MIL-C-22750C Insignia White Epoxy Polymide Enamel. After air drying a minimum of 48 hours, it is cured at 140^{O}F for one hour, which exceeds the maximum internal temperature possible during a maximum heating trajectory.

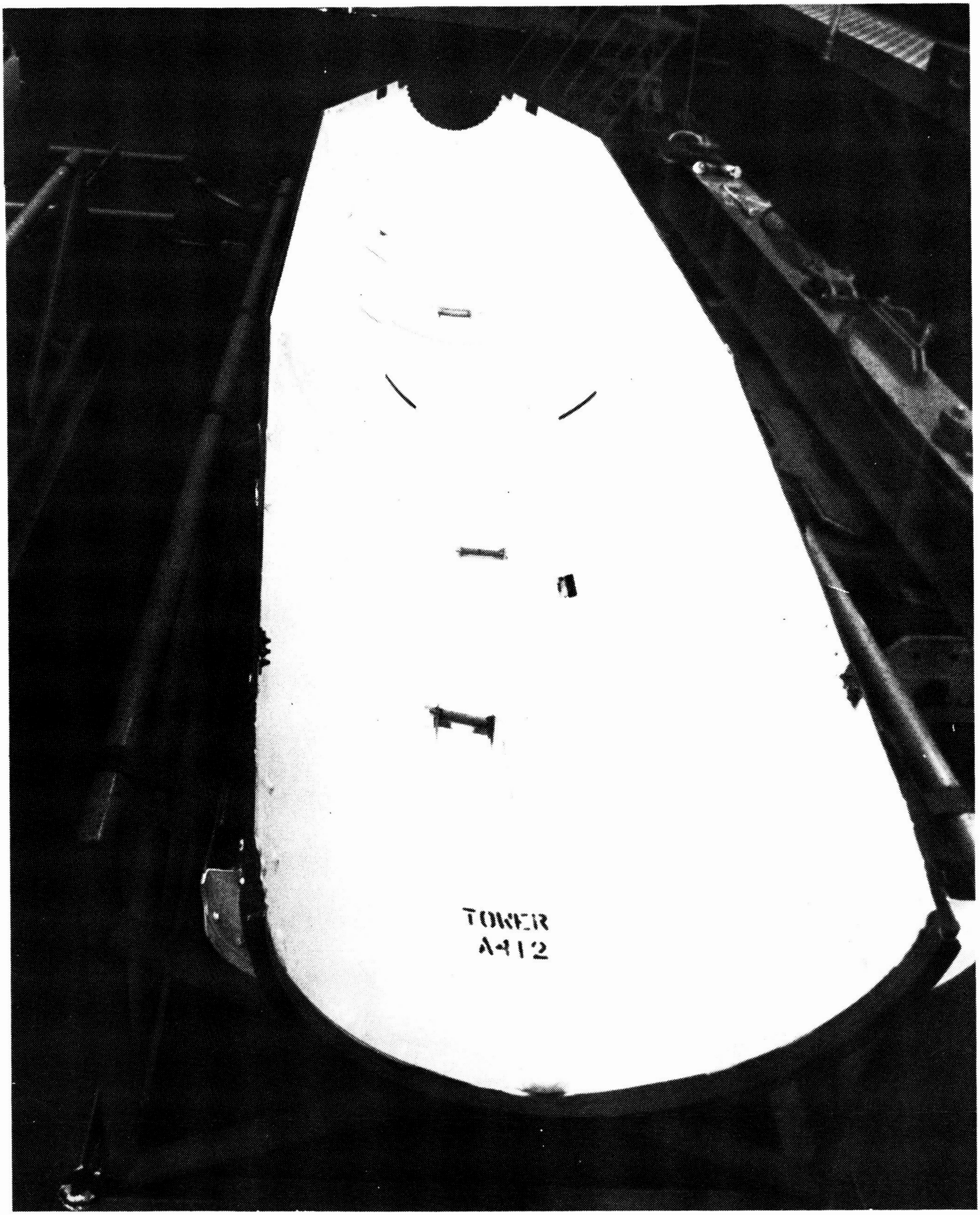

FIGURE V-18 TYPICAL HEATSHIELD ASSEMBLY (SHEET 1 – TOWER SIDE)

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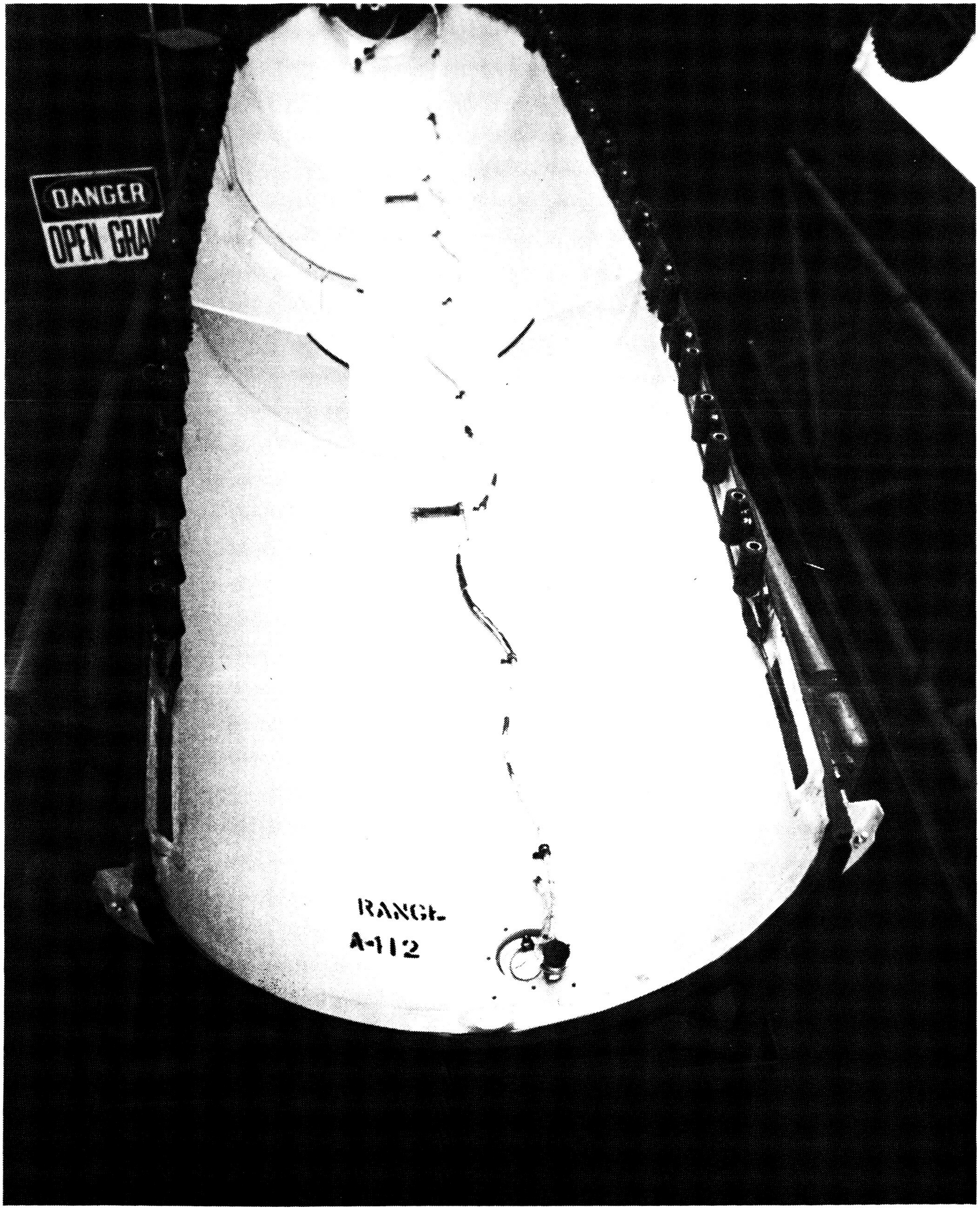

FIGURE V–18 TYPICAL HEATSHIELD ASSEMBLY (SHEET 2 – DOWN RANGE SIDE)

The R-F characteristics of the heatshield are similar to common phenolic resin impregnated glass cloth radomes. R-F transmission through the heatshield is virtually uninhibited for a plane wave at incident angles less than 40 degrees. The transmission loss at X band frequencies has been experimentally determined to be less than 0.5 db for a material sample similar to that used in the Scout.

Heatshield shell restraint is provided by a series of overcenter latches along the separation plaen and the joining clamp at the base. Prior to third stage ignition, a ballistic actuator in the forward end of the heatshield is fired. The latches and clamp are released by drawbars attached to the ballistic actuator bellcranks. Contained springs along the separation plane force the heatshield halves apart.

3.2 PAYLOAD ENVELOPES

The allowable payload usable envelope within the confines of the heatshield are shown in Figures V-19, V-20, V-21, and V-22. The envelopes shown utilize standard adapter sections. Payloads which extend into the area aft of the payload separation plane, or exceed the envelope shown, will require detail coordination with Vought through the Mission Working Group.

A feasibility study has been conducted on extending the 34 inches diameter heatshield to station -55.0. This extension is made possible by adding 15 inches to the cylindrical section of the heatshield, increasing the volume of the payload envelope by approximately 6 cubic feet.

Another feasibility study has been conducted on the 42 inches diameter heatshield to extend it to station -55. This extension makes possible an increase in payload volume of approximately 6.5 cubic feet by adding 10. inches to the cylindrical section.

3.3 HEATSHIELD ACCESS

The number, size, and location of openings in the heatshield vary with the payload requirements and are governed by the structural integrity of the heatshield.

The following is a list of standard size door openings which are available for use by the payload. Other configurations may be considered for special payload needs and would be a topic for discussion by the Mission Working Group.

TYPE	SPRING LOADED DOORS	BOLTED DOORS
SIZE	15.2 X 15.2 Centimeters (6.0 X 6.0 Inches) 20.3 x 20.3 Centimeters (8.0 x 8.0 Inches)	15.2 X 15.2 Centimeters (6.0 X 6.0 Inches) 20.3 X 20.3 Centimeters (8.0 X 8.0 Inches) 25.4 X 25.4 Centimeters (10.0 X 10.0 Inches)

NOTE

In addition to the structural limitation on the location of these doors, there is also a constraint placed on the location of any fly-away umbilical openings. This constraint is dictated by the limits at which the launcher retraction arm can pull. The allowable locations for the payload umbilical doors and standard access doors for each heatshield are shown in Figure V-23, sheets 1, 2, and 3. UMBILICAL DOORS NOT LOCATED IN THE SUGGESTED AREA ARE TO BE AVOIDED AS THEY ENTAIL MODIFICATION OF THE LAUNCH COMPLEX.

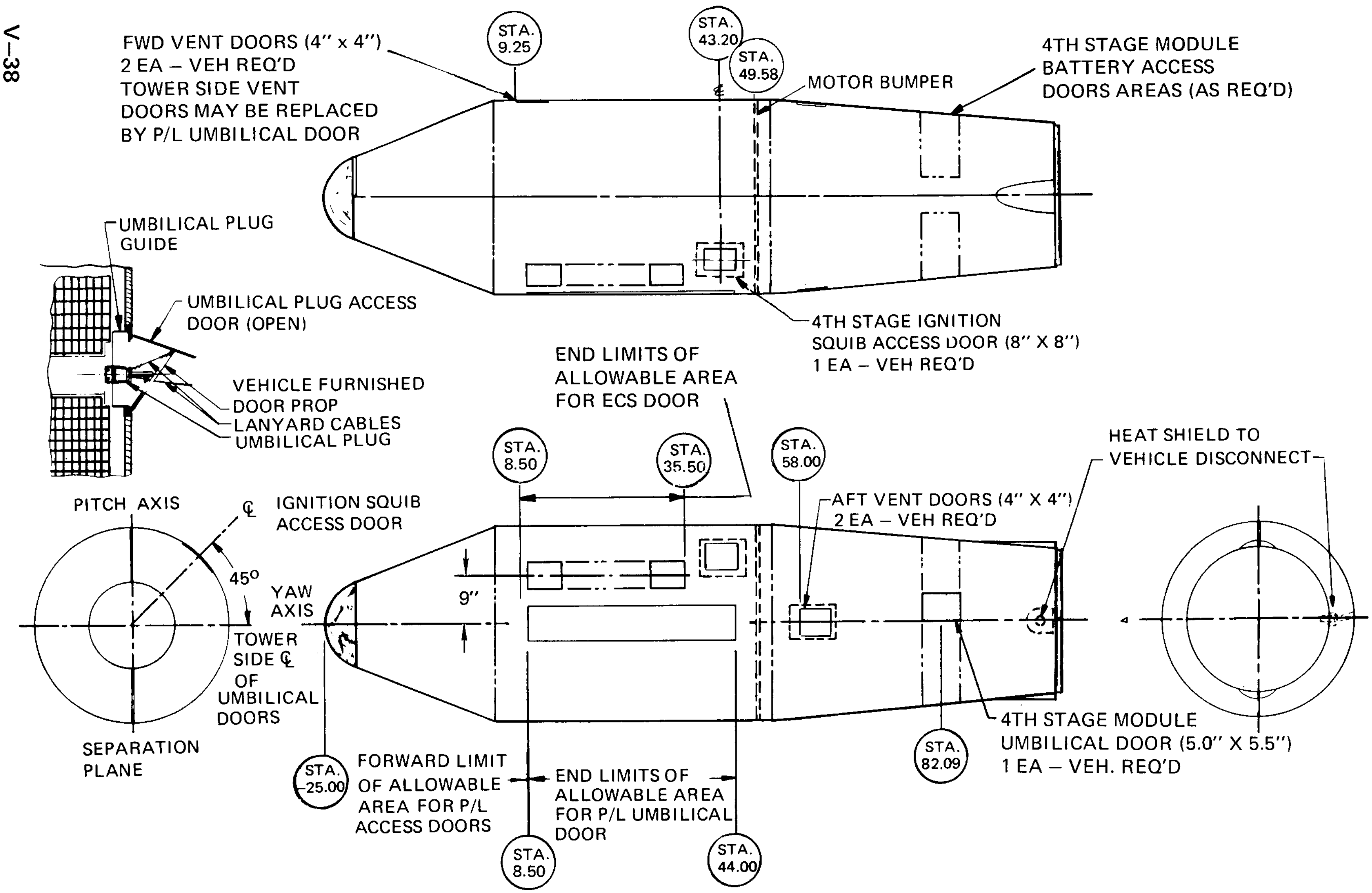

FIGURE V–23 DETAILS AND LOCATION OF PAYLOAD ACCESS DOORS
(SHEET 1 – 34 INCH DIAMETER, (.86 METER DIA.)–25 NOSE STATION)

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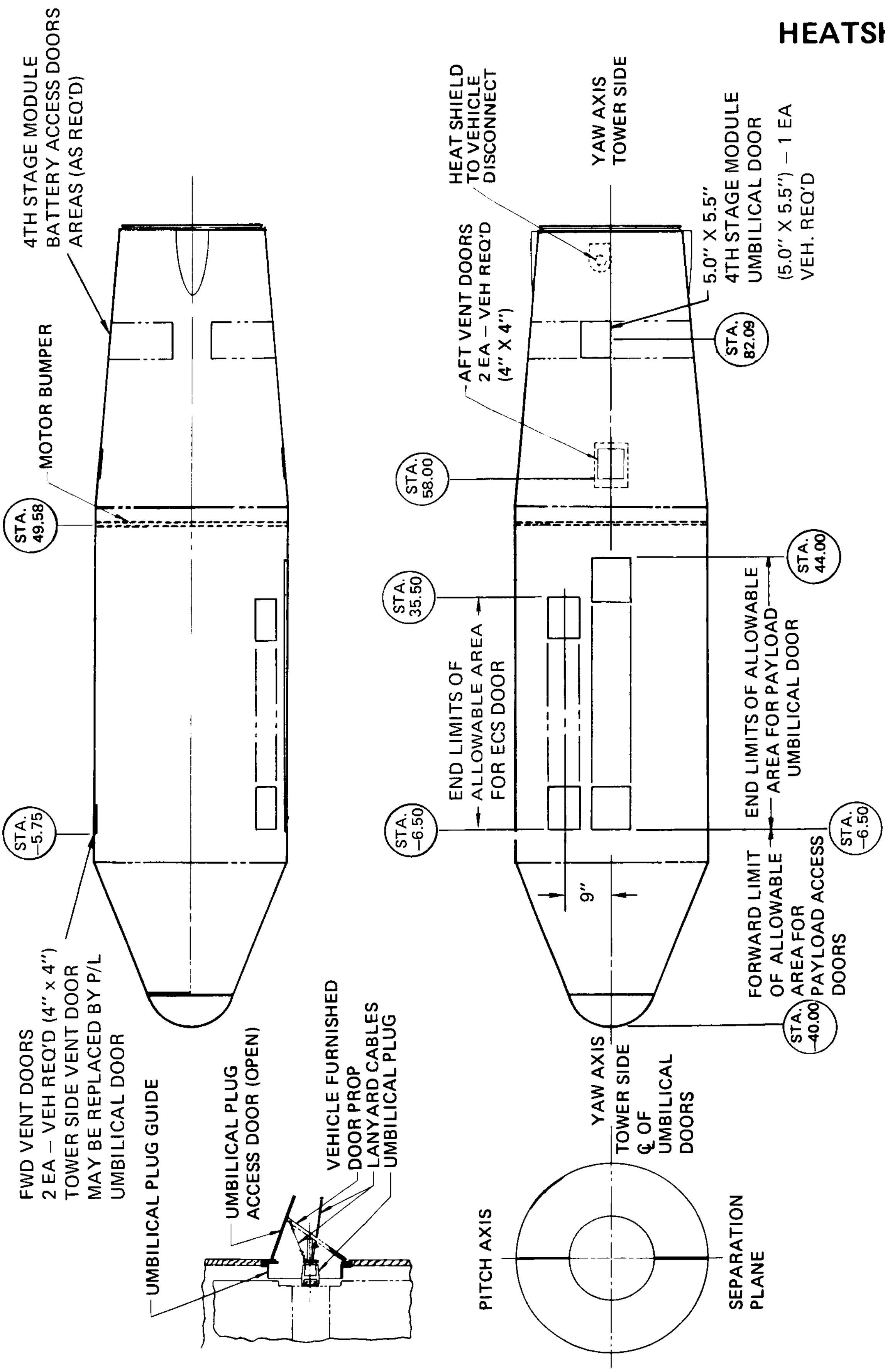


FIGURE V–23 DETAILS AND LOCATION OF PAYLOAD ACCESS DOORS
(SHEET 2 – 34 INCH DIAMETER, (.86 METER DIA.)–40 NOSE STATION)

177

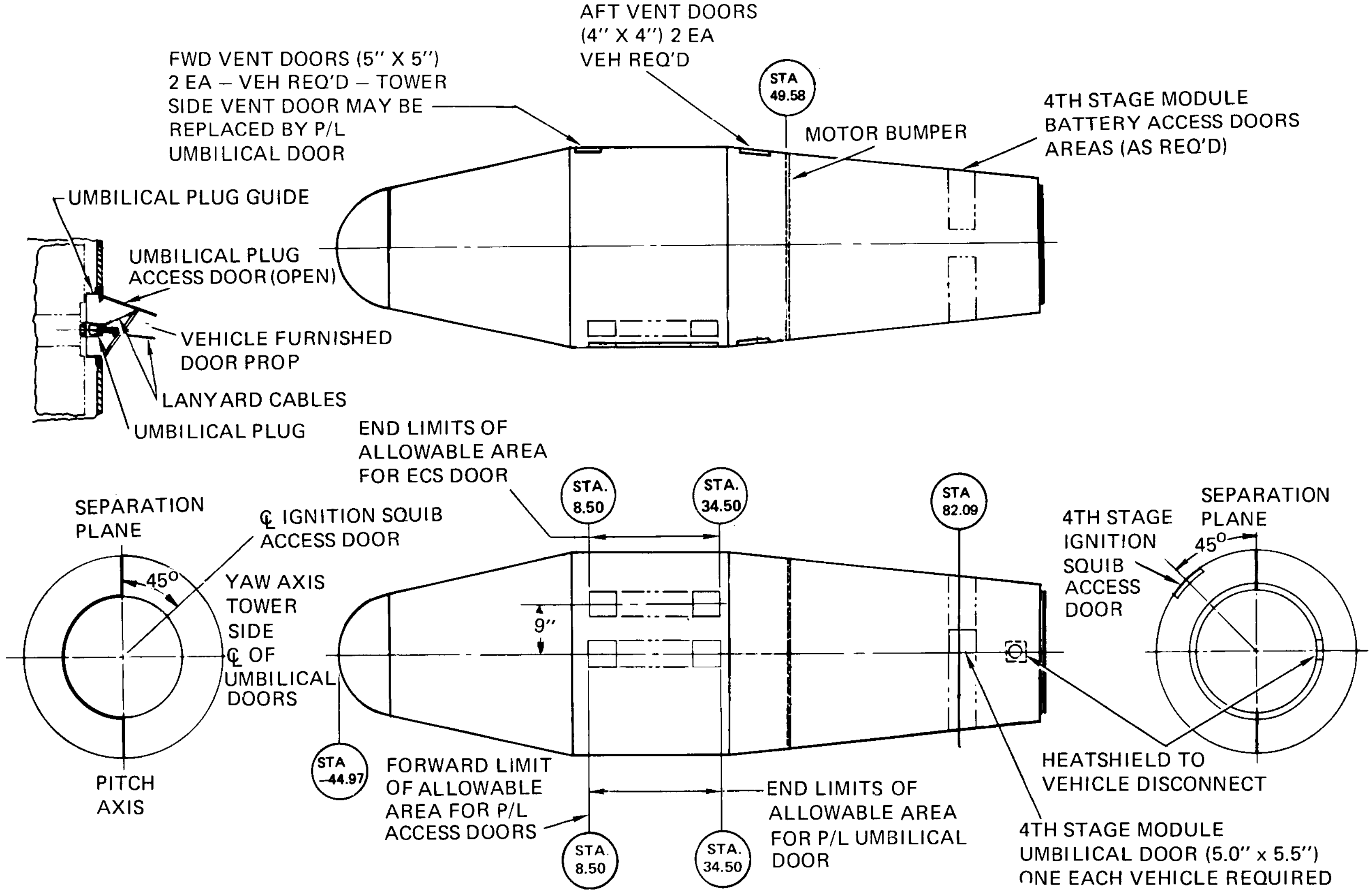


FIGURE V–23 DETAILS AND LOCATION OF PAYLOAD ACCESS DOORS (SHEET 3 – 42 INCH DIAMETER, 1.07 METER DIAMETER – 45 NOSE STATION)

4.0 MECHANICAL DESIGN CRITERIA

This section identifies the vehicle imposed environmental constraints, balance requirements, and other mechanical design criteria.

4.1 FLIGHT ENVIRONMENT

The environmental conditions to which the spacecraft may be subjected during the powered launch phase are described in the ensuing paragraphs.

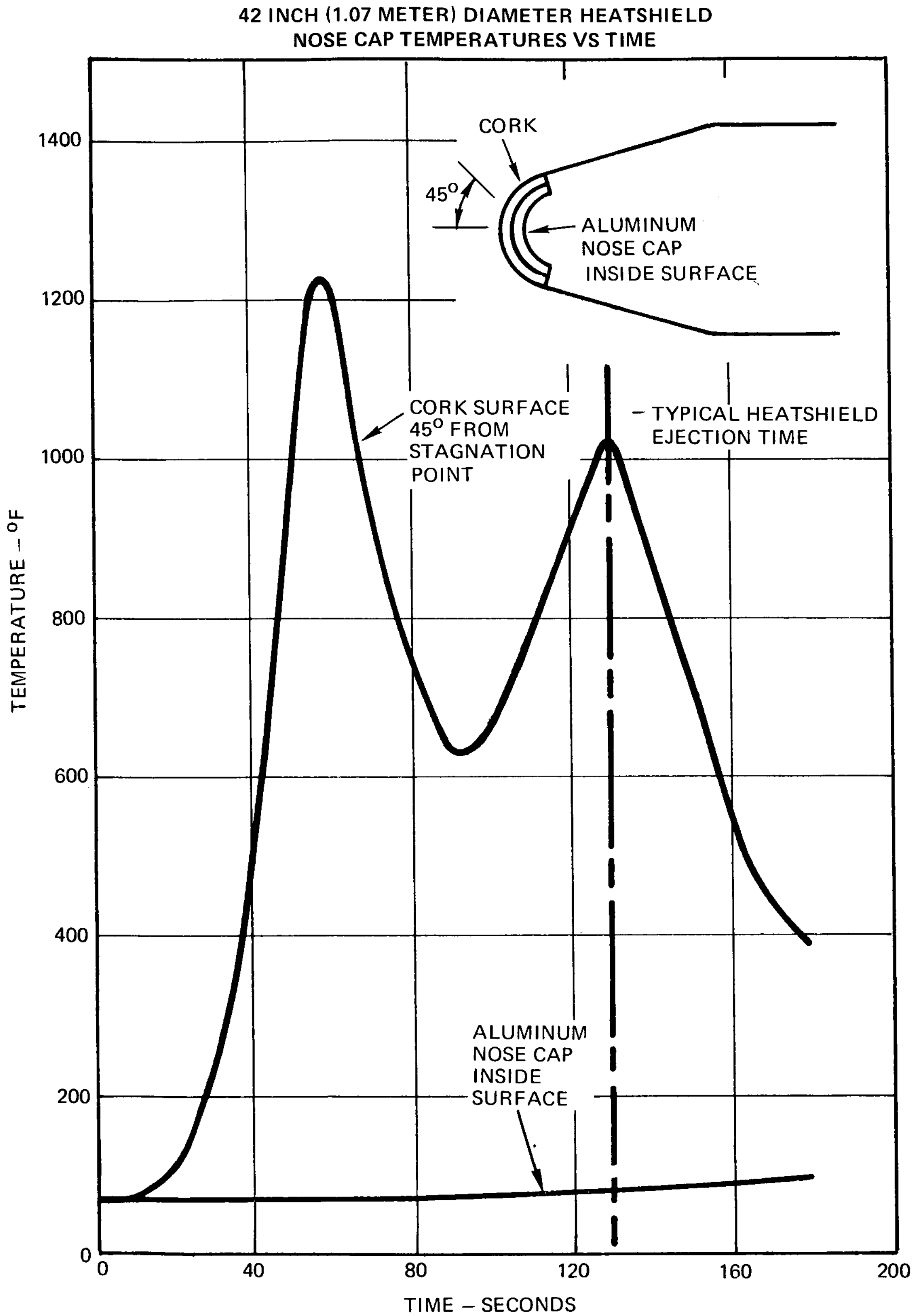
4.1.1 Heatshield Temperature

Figure V-24 presents the nose cap temperature versus time for the 42 inch diameter heatshield. Figure V-25 presents the heatshield conical and cylindrical section outside and inside temperatures versus time.

Figure V-26 presents the heatshield nose cap stagnation point temperature versus time for the 34 inch diameter heatshield. The heatshield conical and cylindrical section outside and inside temperatures versus time are presented in Figure V-27.

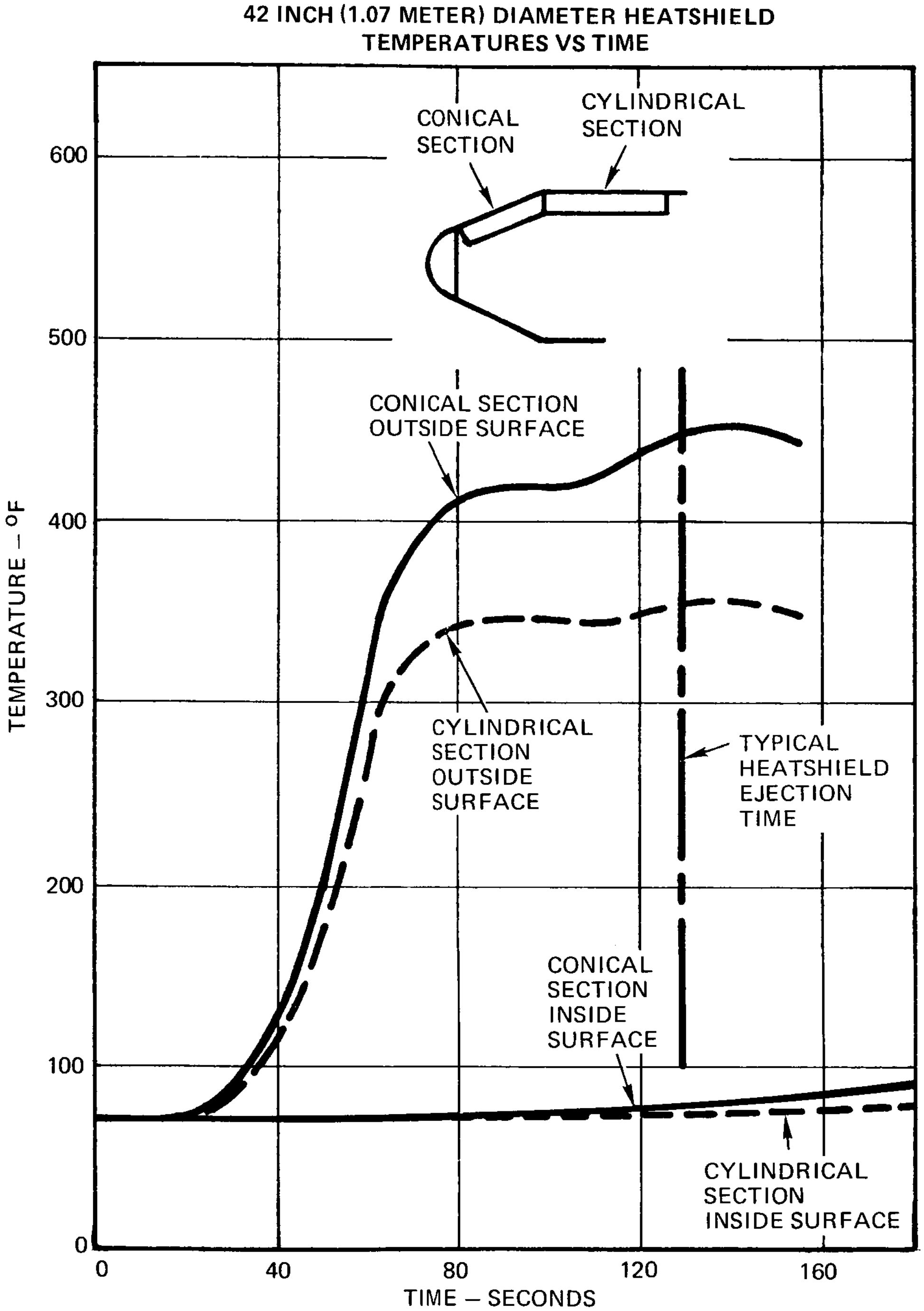
The average emissivity of the heatshield inner surface as manufactured = 0.88. The outer surface of the heatshield is painted white. This information is given to aid the Payload Agency in computing the thermodynamic environment for the payload compartment if desired. Absorptivity of external surface = 0.50. Emissivity of external surface = 0.80.

The temperature rise of the payload will be a function of its proximity, surface emissivity, and the amount of payload insulation placed between it and the heatshield.



NOTE: TEMPERATURES ARE BASED ON A MAXIMUM HEATING TRAJECTORY

FIGURE V–24 NOSE CAP TEMPERATURE

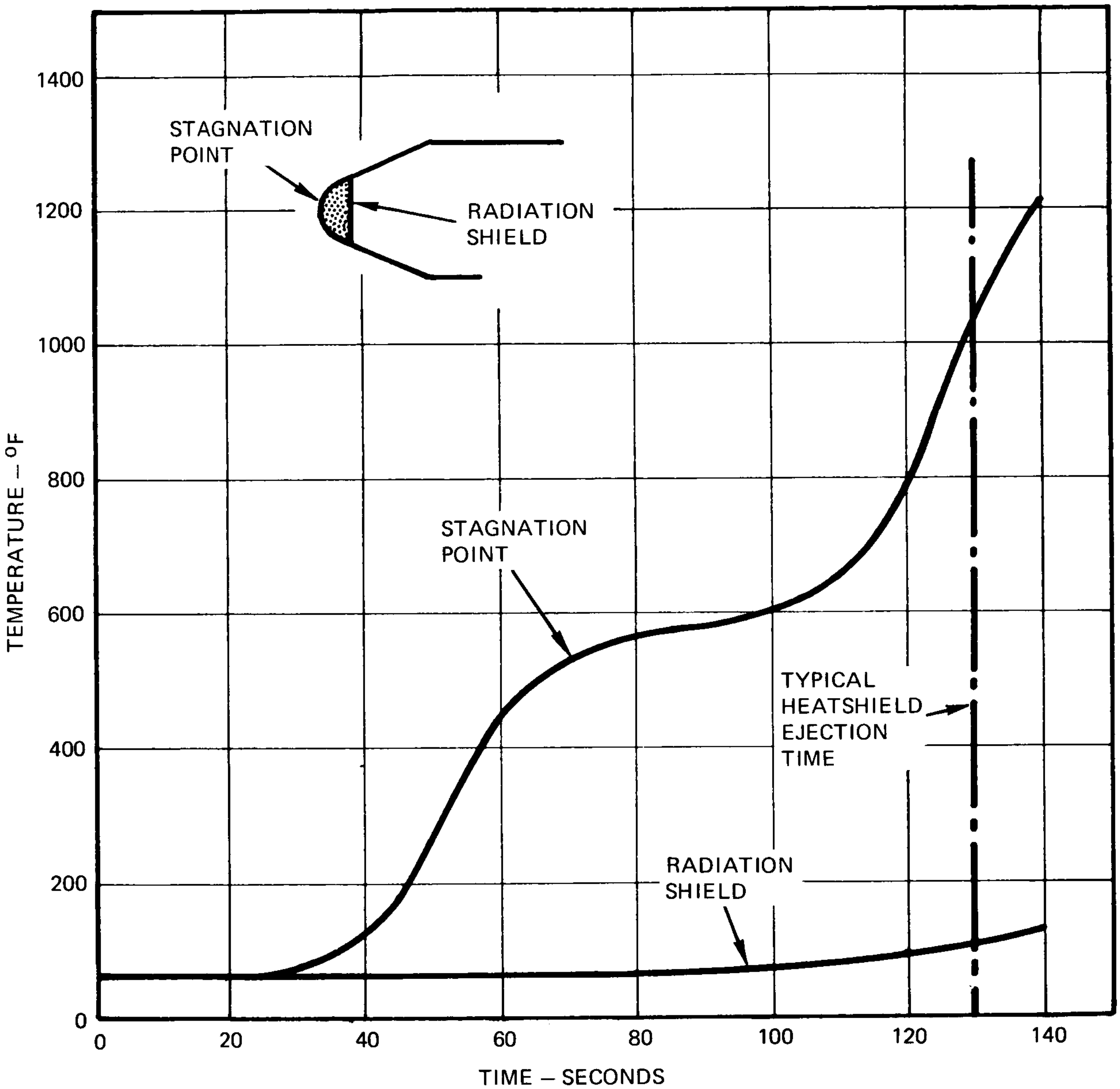


NOTE: TEMPERATURES ARE BASED ON A MAXIMUM HEATING TRAJECTORY

FIGURE V–25 HEATSHIELD OUTSIDE AND INSIDE TEMPERATURES

34 INCH, (.86 METER), DIAMETER HEATSHIELD

NOSE CAP STAGNATION POINT AND RADIATION SHIELD
TEMPERATURES VS TIME

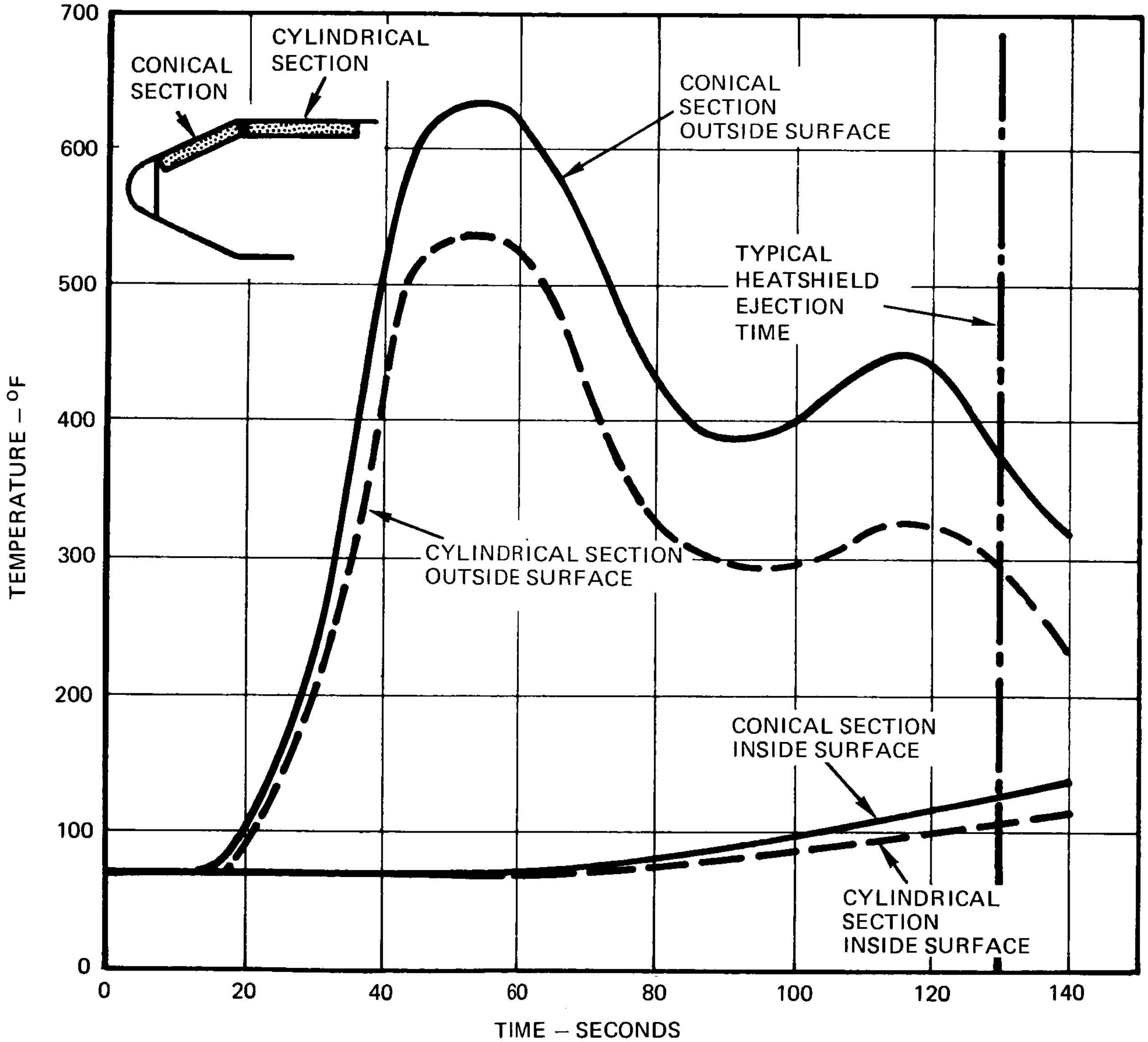


NOTE: TEMPERATURES ARE BASED ON MAXIMUM HEATING TRAJECTORY

FIGURE V–26 NOSE CAP STAGNATION POINT AND RADIATION SHIELD TEMPERATURES

34 INCH (.86 METER) DIAMETER HEATSHIELD

HEATSHIELD
OUTSIDE AND INSIDE TEMPERATURES
VS TIME
CONICAL AND CYLINDRICAL SECTIONS



NOTE: TEMPERATURES ARE BASED ON MAXIMUM HEATING TRAJECTORY

FIGURE V–27 HEATSHIELD OUTSIDE AND INSIDE TEMPERATURES

4.1.2 Heatshield Pressure

Venting of the heatshield during flight is accomplished by four spring-loaded vent doors installed in the heatshield. Two doors are located on the forward end of the cylindrical section of the heatshield, and the other two are located on the forward end of the aft conical section. Additional venting area is provided by the spring loaded umbilical doors for the Fourth Stage Module Telemetry/Separation Systems and payload. Figure V–28 presents the ambient pressure inside the heatshields versus flight time for 34 and 42 inch heatshields.

4.1.3 Fourth Stage Motor Temperatures

The typical external temperature curves of the fourth stage motor due to motor operation are shown in Figure V-29, and represent worst case ALTAIR IIIA temperatures on the forward motor dome (T_D) and middle motor case (T_C) from ignition to 600 seconds. These curves represent case radial locations where the most severe heating occurs. An optional thermal blanket is available for the motor dome in the event that lower temperatures in this area are required. The temperature effect at a point on "E" section with (T_E') and without (T_E) the thermal blanket installed is illustrated in Figure V–29.

4.1.4 Spacecraft Heating Rates

Spacecraft aerodynamic heating rates after heatshield ejection are presented in Figure V–30 for a range of air relative velocities and heatshield ejection altitudes. These heating rates are for a surface normal to the airstream and were computed for free molecular flow. Effects of other surface orientations and continuum flow will depend upon specific spacecraft configurations.

4.1.5 Fourth Stage Residual Thrust

A typical residual thrust curve for the ALTAIR IIIA fourth stage motor is shown in Figure V–31.

4.1.6 Fourth Stage Half-Cone Angle

Fourth stage half-cone angle data, based on flight experience, is presented in Tabel V–1. These data show the mean and three-sigma values and the standard deviation of half-cone angle at fourth stage separation, ignition and burnout. Figure V–32 presents the predicted three-sigma value of half cone angle at fourth stage ignition as a function of spin rate.

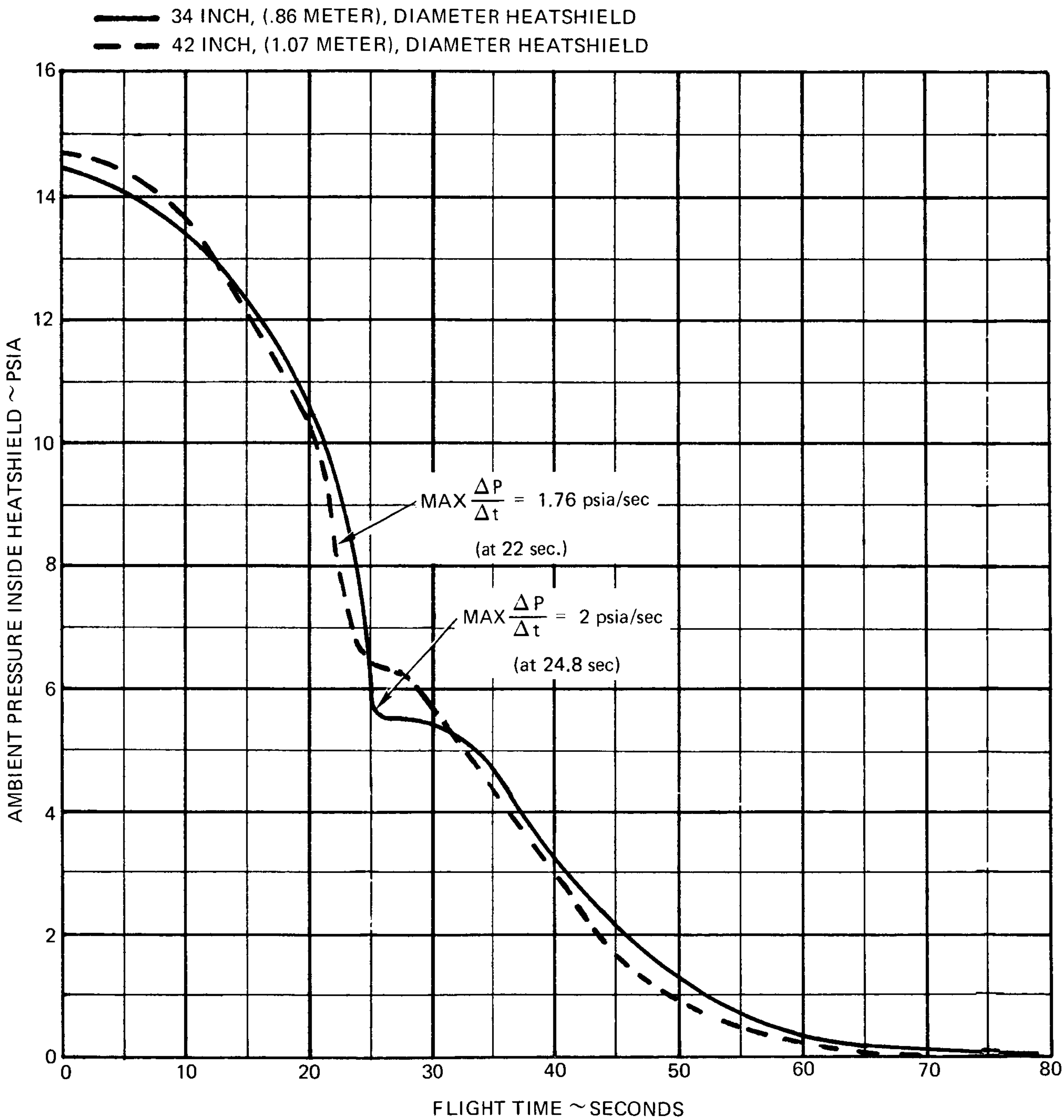
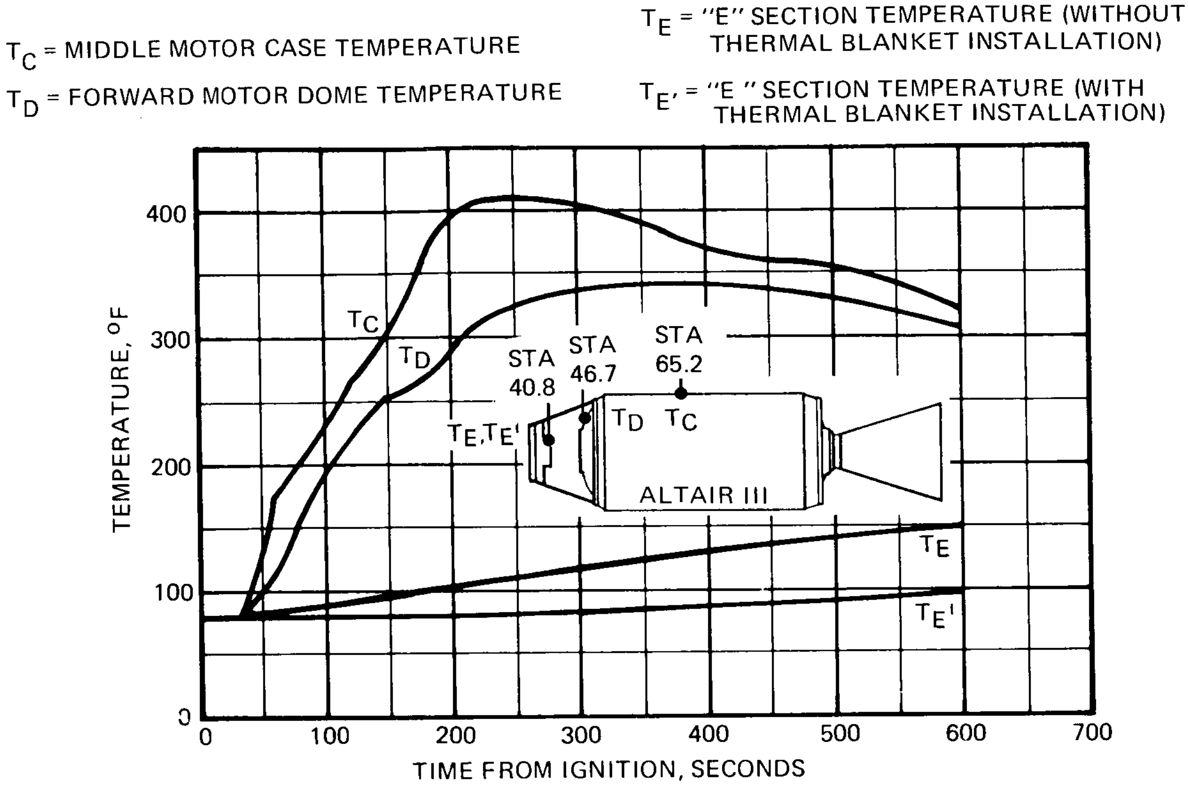


FIGURE V–28 AMBIENT PRESSURE INSIDE HEATSHIELD



ALTAIR III ROCKET MOTOR

FIGURE V–29 FOURTH STAGE ROCKET MOTOR PEAK SKIN TEMPERATURE VERSUS TIME AFTER IGNITION

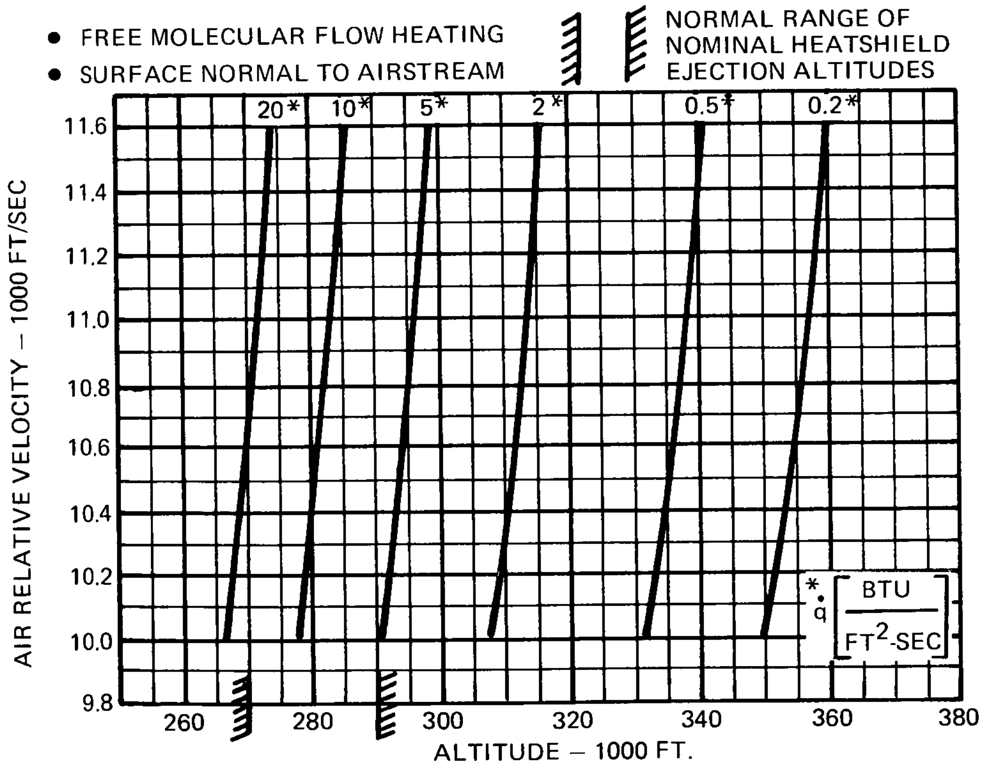


FIGURE V–30 SPACECRAFT HEATING RATES AFTER HEATSHIELD EJECTION

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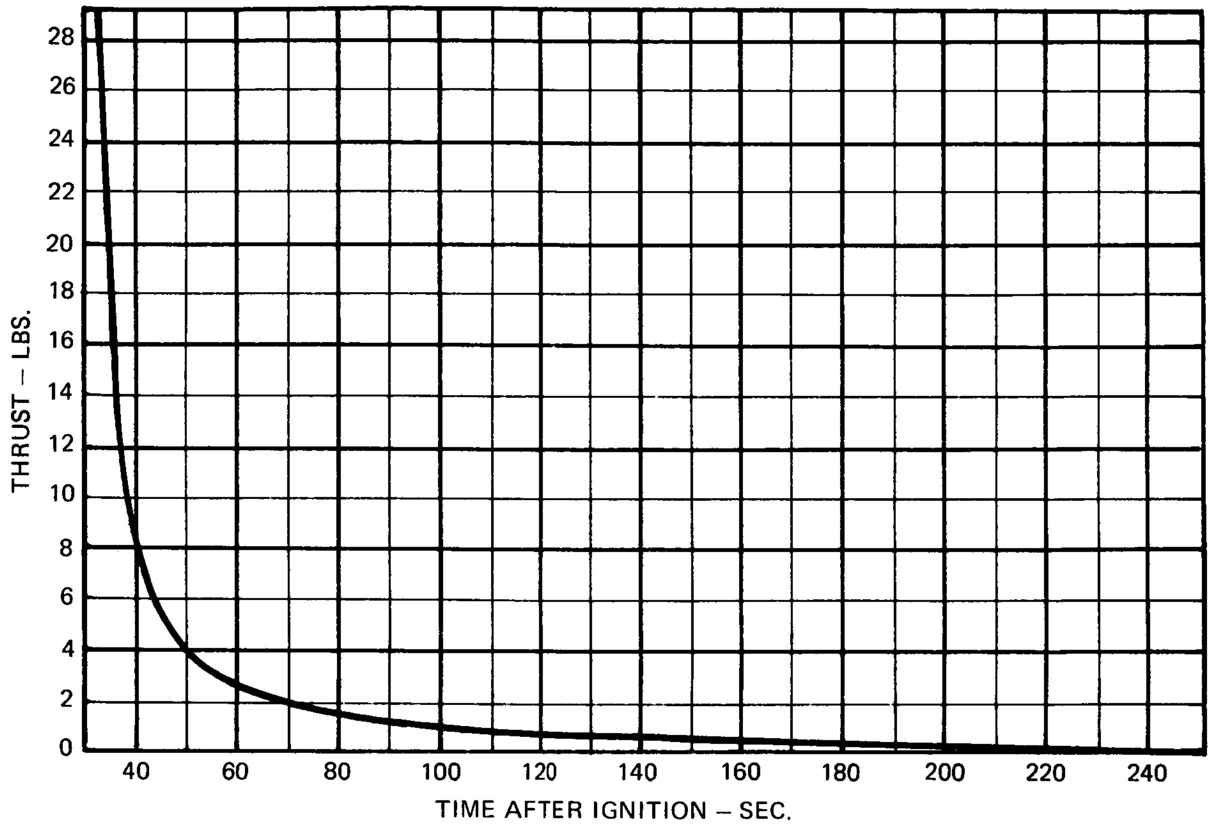

FIGURE V–31 ALTAIR IIIA RESIDUAL THRUST VS TIME

TABLE V–1 OBSERVED FOURTH STAGE HALF-CONE ANGLE

FOR SPIN RATE RANGE OF 136 TO 184 RPM AVERAGE SPIN RATE OF 158 RPM

EVENT	MEAN (DEGREES)	(+3) SIGMA VALUE (DEGREES)	STANDARD OF DEVIATION (DEGREES)
4TH STAGE SEPARATION	0.65	1.79	0.38
4TH STAGE IGNITION	1.40	3.66	0.75
4TH STAGE BURNOUT	0.53	1.14	0.20

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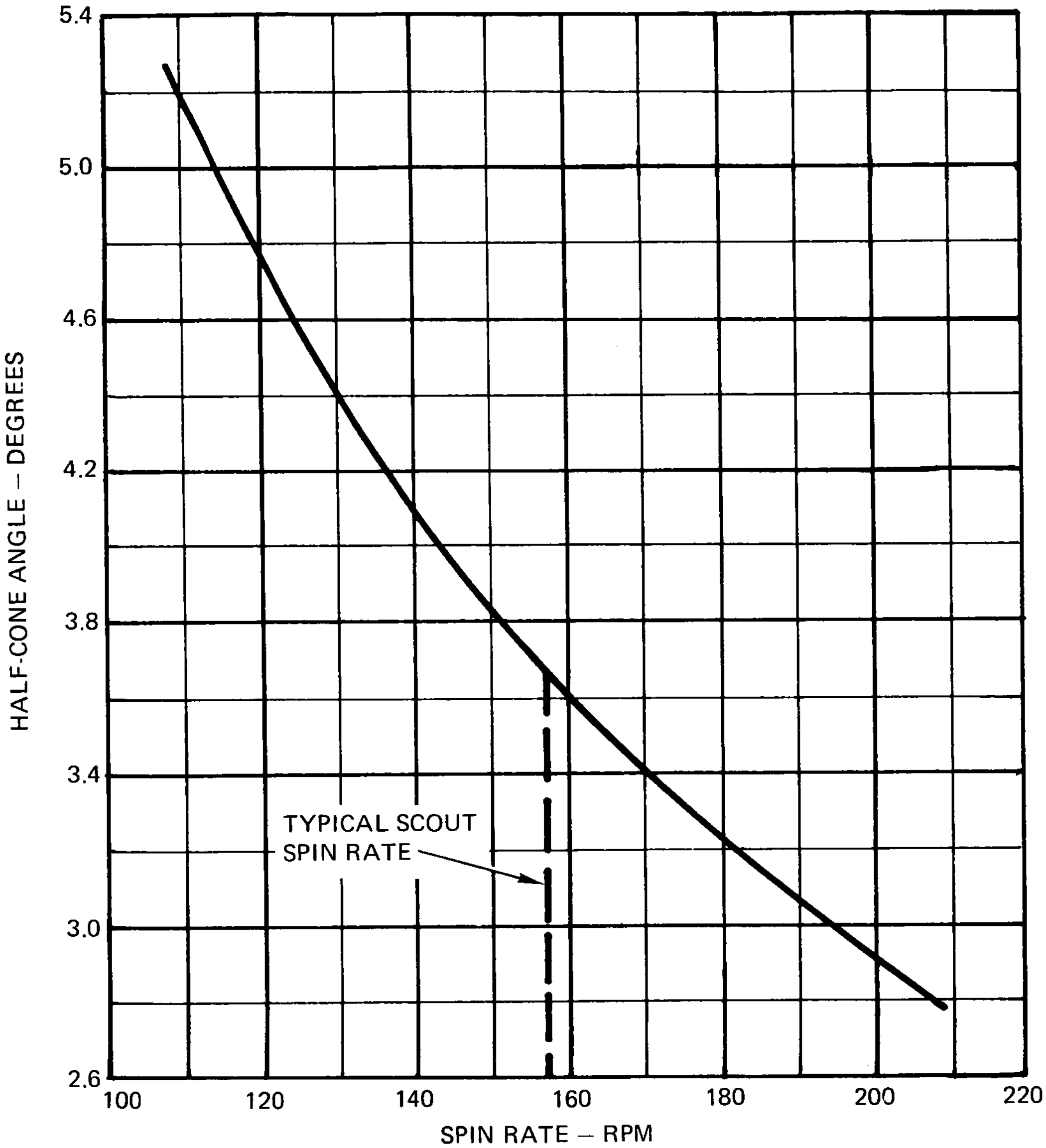

FIGURE V–32 PREDICTED THREE-SIGMA HALF-CONE ANGLE AT FOURTH STAGE IGNITION VS SPIN RATE

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4.1.7 Fourth Stage Case Outgassing

Test panels for fourth stage outgassing determination were cut from cylindrical test specimens that represented the Altair IIIA fourth stage Scout motor. The test panels were subjected to a simulated temperature-time profile in vacuum with various payload sensors and optical elements positioned in close proximity to the specimens. Thermogravimetric analysis indicated that test specimen weight loss during the flight temperature profile (75^{o} to $400^{o}F$) was approximately 1.0%. Residual gas analysis indicates that some decomposition and outgassing products were given off beginning at a temperature of 212^{o}. Analysis indicates that these outgassing products are caused by the Nadic methyl anhydride of the resin formulation and consist primarily of H_2O, CO, CO_2 and ring fragments.

4.1.8 Steady-State Acceleration

The maximum lateral acceleration is approximately 0.4g experienced during second stage coast and results from the operation of the Reaction Control System. Maximum longitudinal acceleration experienced by a Scout payload will occur during either third or fourth stage thrust, depending on the weight of the payload. The variation of the acceleration with third stage burning time for various payload weights is shown in Figure V–33. The variation of the acceleration with fourth stage burning is presented in Figure V–34.

4.1.9 Shock Environment

To limit the response shock loads induced by stage ignitions, the payload structure should be designed to produce a first cantilevered bending mode of the payload between 15 and 30 Hz. For payloads unable to meet this criteria, it is necessary to coordinate the structural design closely with the Scout Project Office so that appropriate analyses can be performed, if necessary, to define loading conditions.

Maximum composite shock spectra envelopes obtained from analyses of measured flight data are presented for the longitudinal axis in Figure V–35 and for the transverse axes in Figure V–36.

Transient shock at spacecraft separation is induced by two sources: (1) firing of pyrotechnic devices, and (2) release of strain energy stored in the separation clamp. The major portion of the separation shock is generated by the second of these sources, release of the clamp energy. Measurements made during separation testing of Scout transition sections indicate that components located near the separation plane experience high acceleration levels at high frequency for a very brief time period. Measurements made at a distance of 6 to 10 inches from the separation plane indicate negligible acceleration levels. Therefore, the location of sensitive components near the separation plane should be carefully examined.

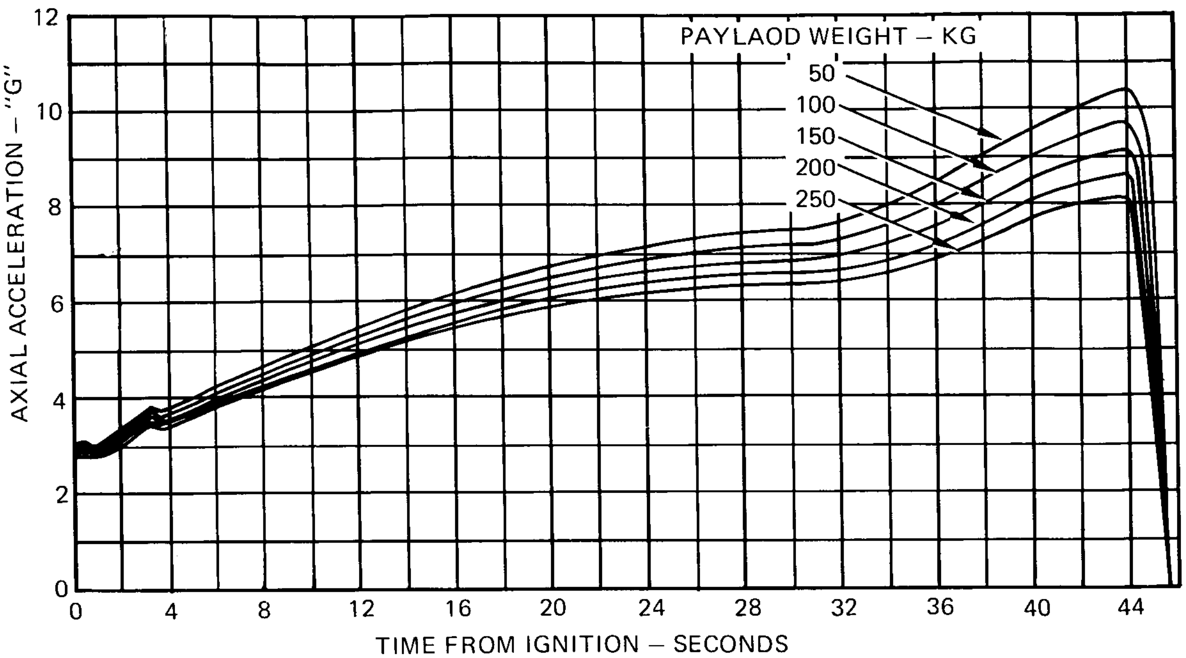
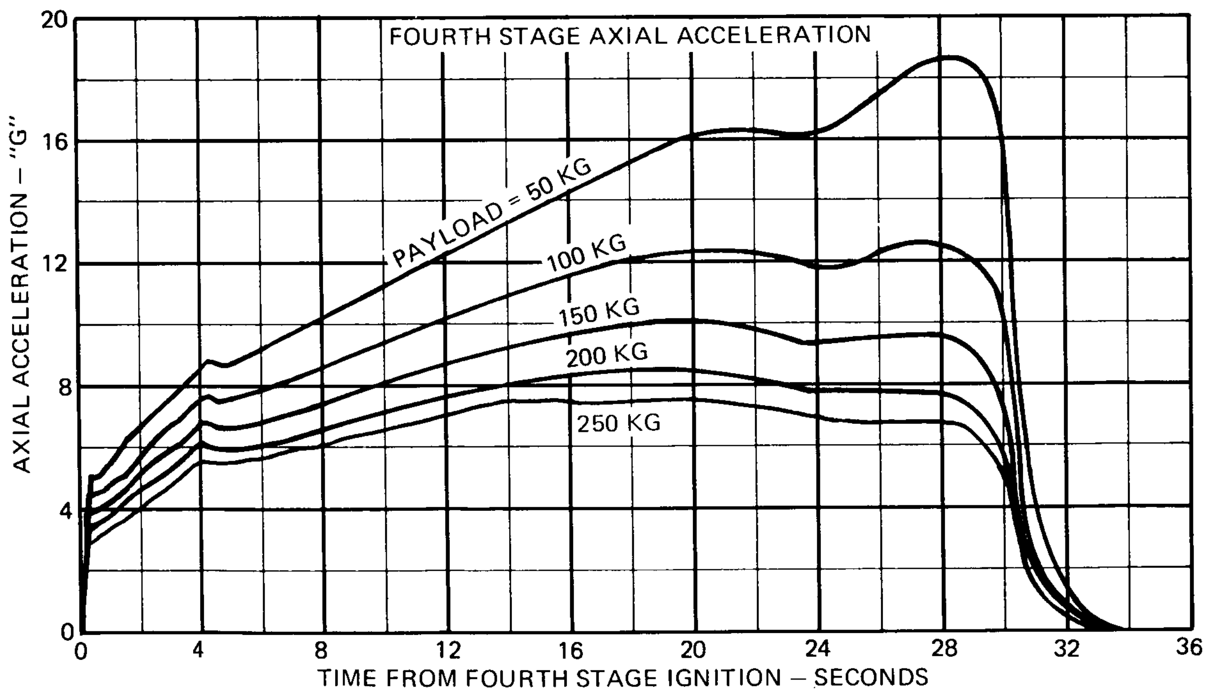


FIGURE V–33 AXIAL ACCELERATION DURING THIRD STAGE THRUSTING FOR VARIOUS PAYLOADS



NOTE: THREE SIGMA VALUES OF ACCELERATION VARY FROM PLUS OR MINUS 0.8 "G", FOR A 50 Kg PAYLOAD, TO PLUS OR MINUS 0.34 "G", FOR A 200 Kg PAYLOAD.

FIGURE V–34 AXIAL ACCELERATION DURING FOURTH STAGE THRUSTING FOR VARIOUS PAYLOADS

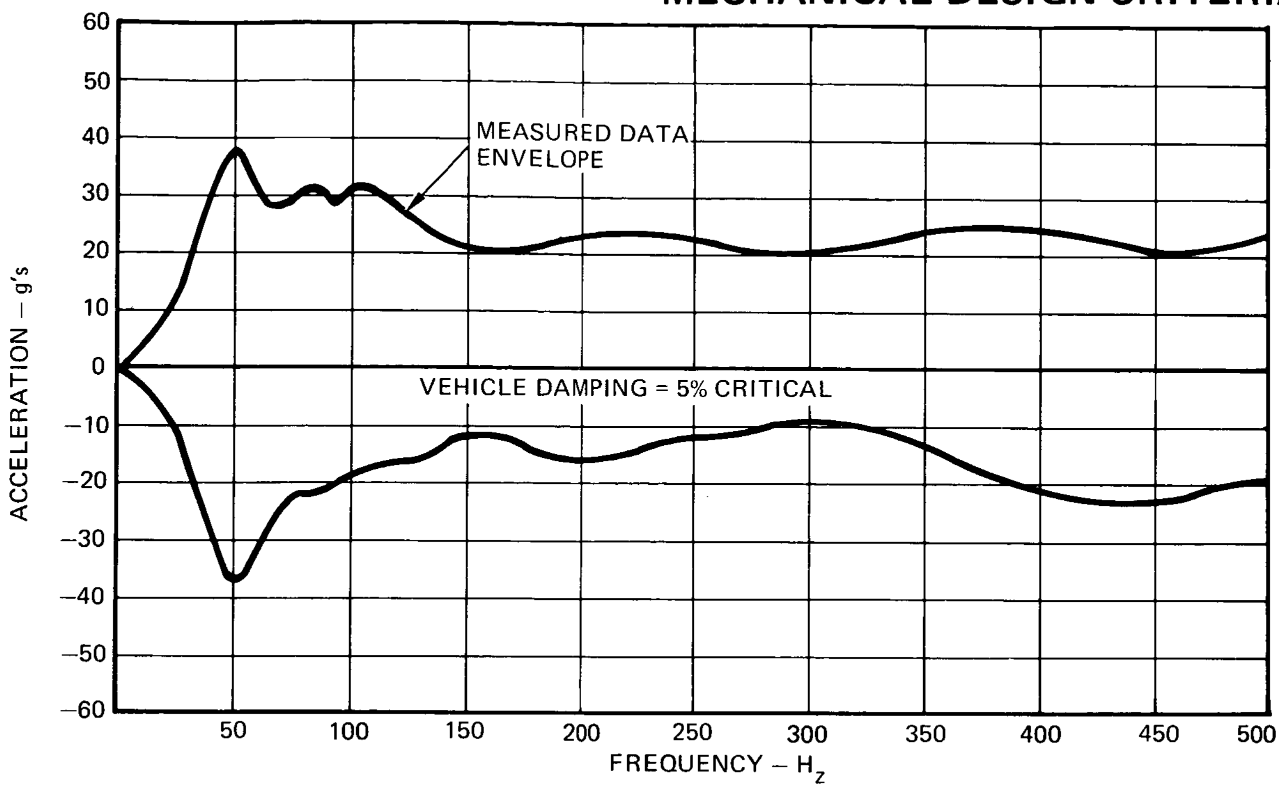


FIGURE V–35 OBSERVED ACCELERATION SHOCK SPECTRUM FOR SCOUT TRANSITION SECTION--LONGITUDINAL AXIS

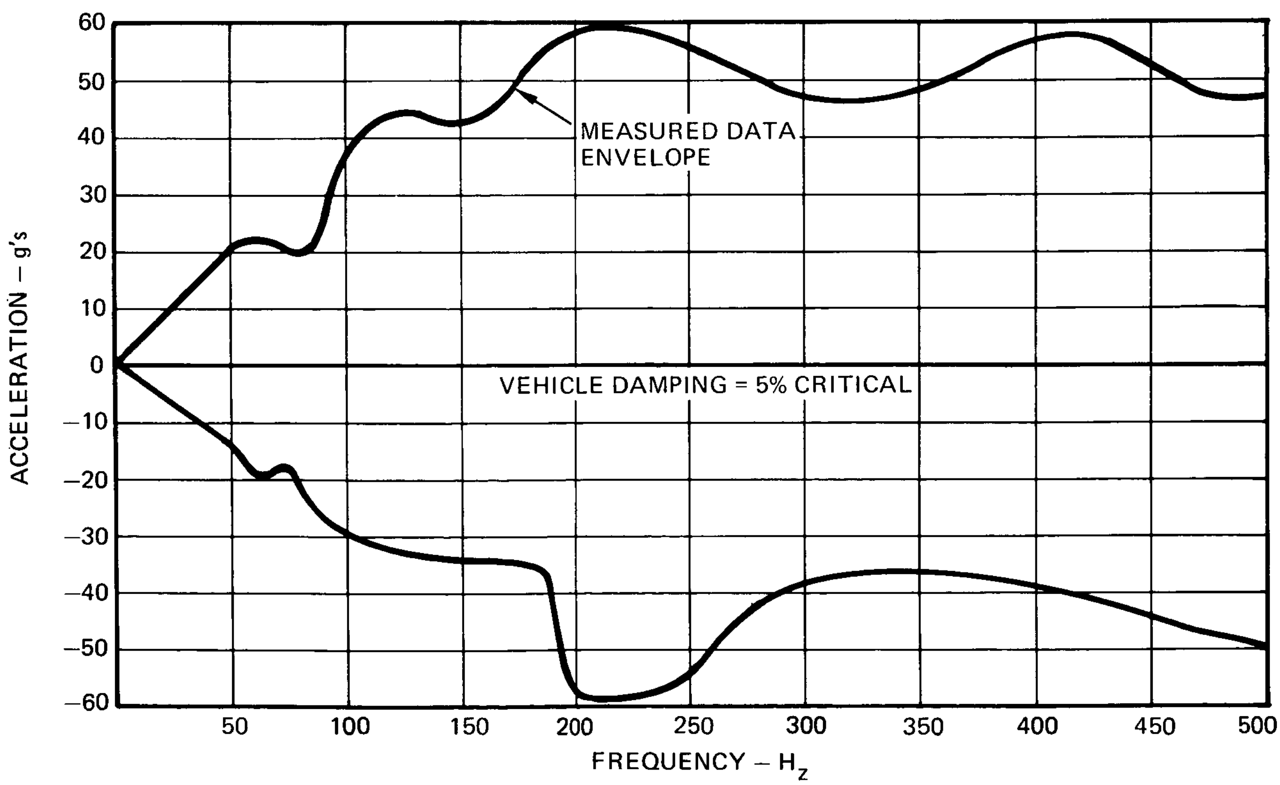


FIGURE V–36 OBSERVED ACCELERATION SHOCK SPECTRUM FOR SCOUT TRANSITION SECTION--TRANSVERSE AXES

4.1.10
Random Vibration Environment

The random vibration spectrum of 95% confidence derived from measured flight data at the time of maximum dynamic pressure is presented in Figure V–37.

4.1.11
Acoustic Environment

1/3 octave band sound pressure levels at the time of maximum dynamic pressure for a typical Scout vehicle are presented in Figure V–38. These levels represent a composite of several vehicles. Figures V–39 and V–40 present measured sound pressure levels both at lift-off and in the transonic region, respectively, for a typical Scout flight.

4.1.12
Dynamic Balancing and Spin Rates

The following paragraphs define the acceptable balance limits for the payload and the nominal spin rates which the payload can encounter in flight.

Special consideration should be given during the design phase of the payload to replacement of major weight components such as batteries, power supplies, tape recorders, etc. Close tolerance mounting provisions should be made, wherever possible, to prevent adverse effect to the balance by replacement of a component. The use of floating type nut plates should be avoided.

Spin balance operations are described in the Operations Section of Chapter IV. Special attention should be given to spacecraft which contain energy dissipating systems such as unbaffled or partially filled tanks of liquid, and vibration isolation systems. Before beginning spin balance operations, the effect of these energy dissipating systems should be reduced to a minimum by baffling or otherwise eliminating liquid free surface and by appropriate blocking out of vibration isolation systems. Balance weights, as required, are installed at the attach points provided for that purpose on the fourth stage motor forward attach flange and on the upper D-section.

Should the design of the payload or its adapter be such that the attach points on the motor forward attach flange are not accessible for balance weight installation, then provisions must be made for attaching the required balance weights to the payload equipment.

4.1.12.1 Dynamic Balancing Data – Fourth Stage

The acceptable mass unbalance of the payload with respect to the axis passing through the center of the support ring normal to the plane of the support ring is as follows:

Dynamic Unbalance: 36,500 gm-cm^2 (200 oz-in^2)

Static Unbalance: 800 gm-cm (12 oz-in)

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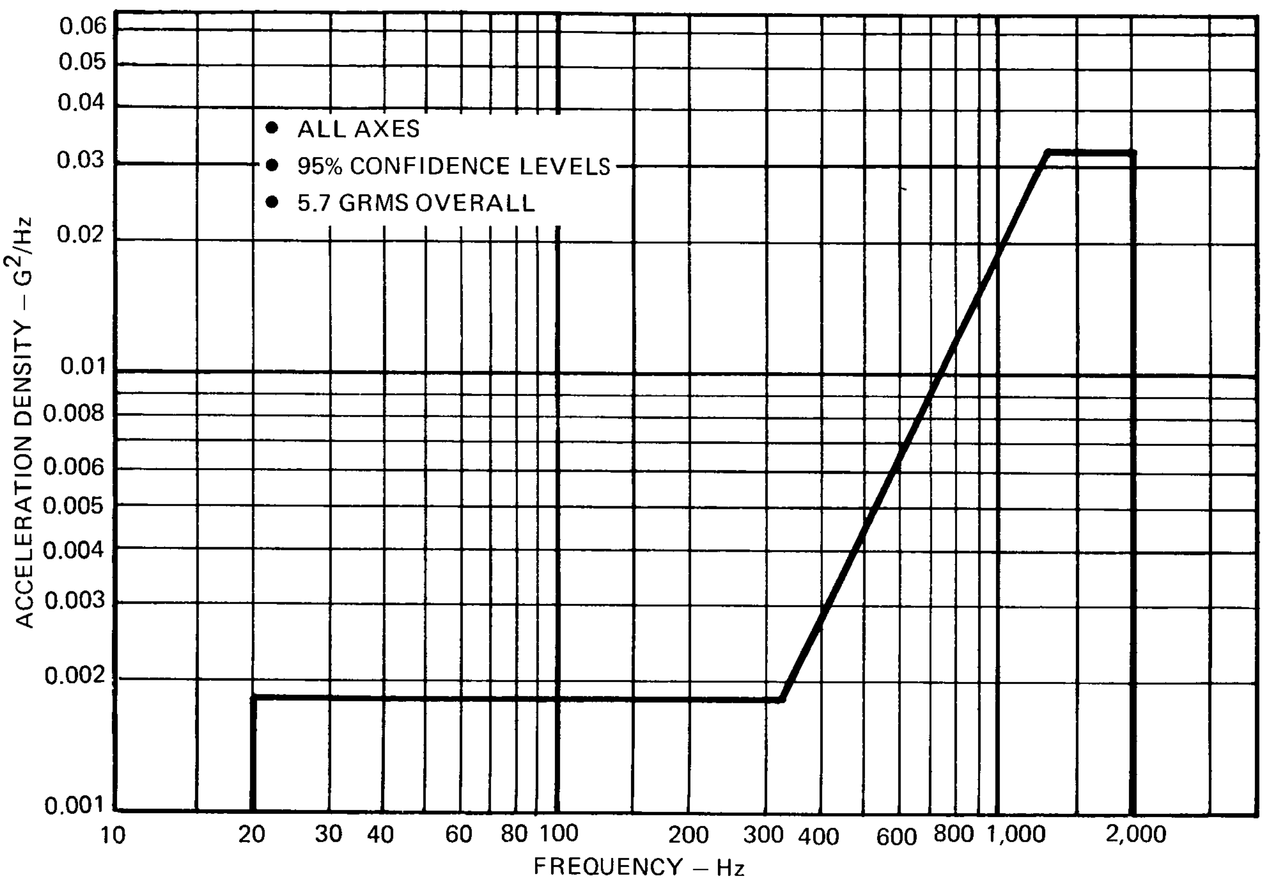

FIGURE V–37 OBSERVED RANDOM VIBRATION ENVIRONMENT–ALL AXES

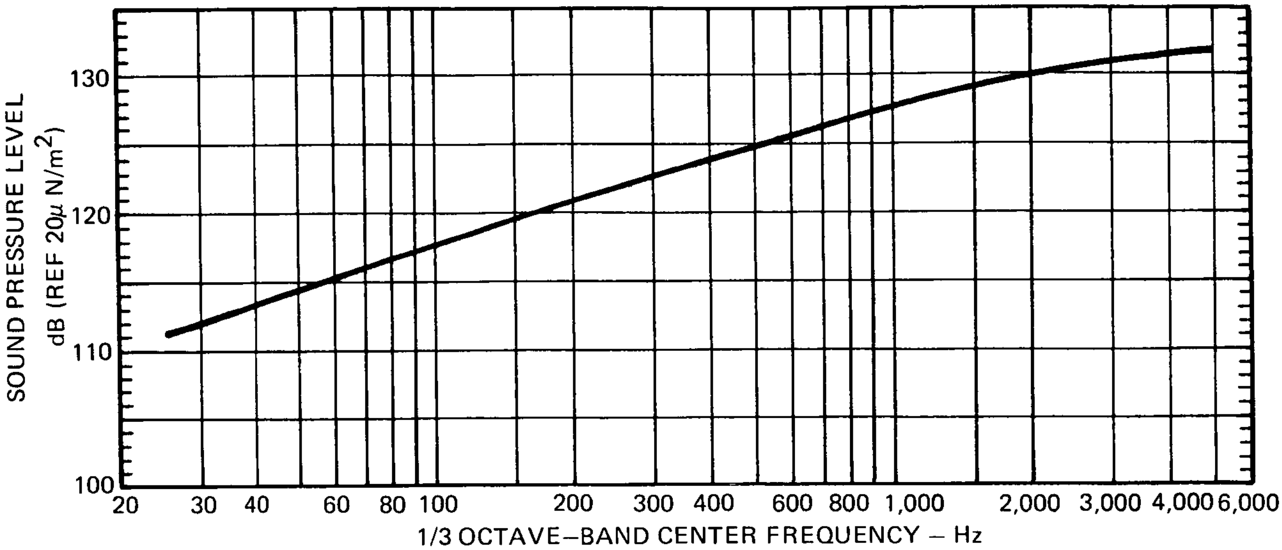

FIGURE V–38 1/3 OCTAVE–BAND SOUND PRESSURE LEVELS AT MAXIMUM DYNAMIC PRESSURE

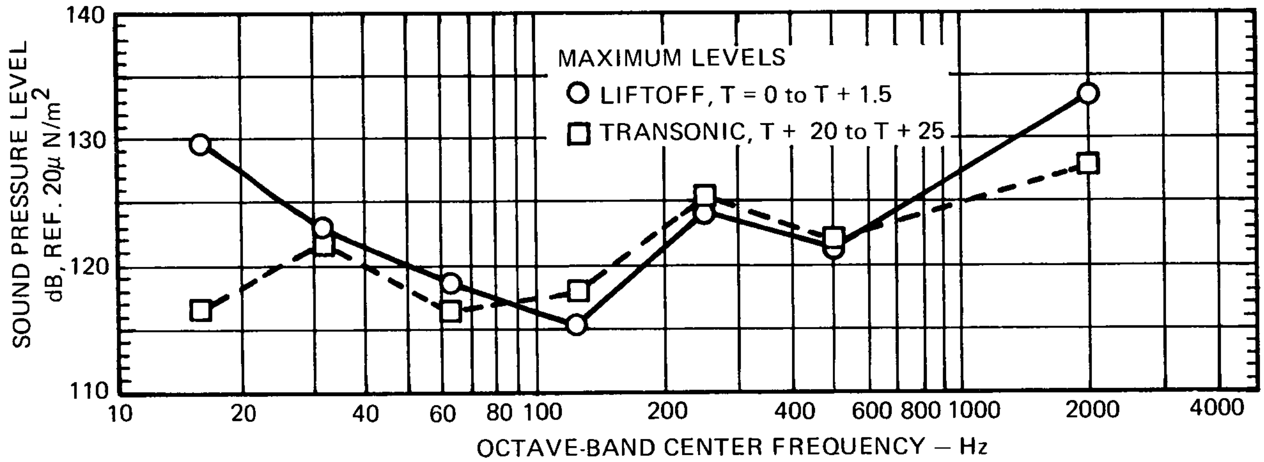

FIGURE V–39 IN-FLIGHT OCTAVE BAND SOUND PRESSURE LEVELS

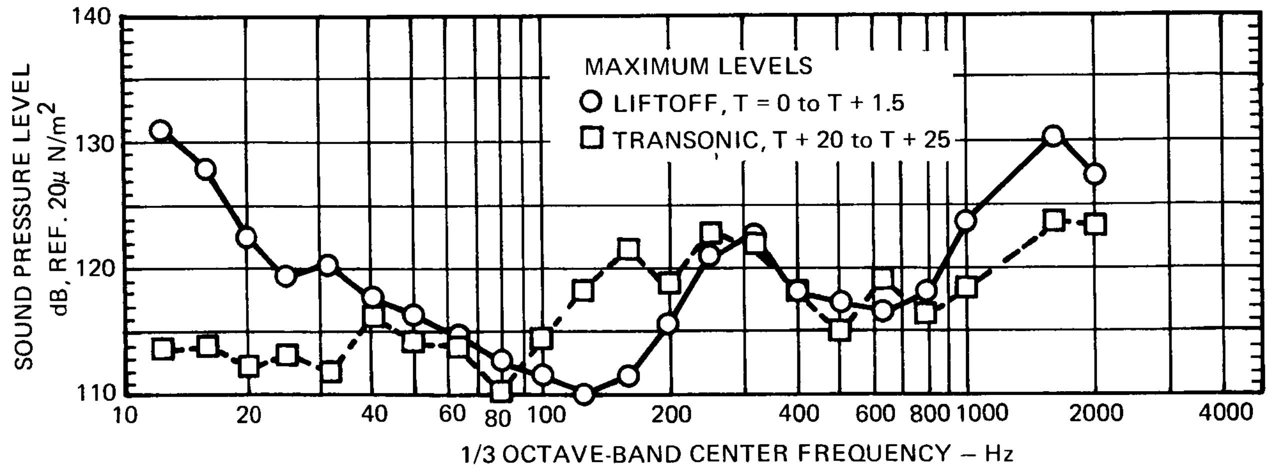

FIGURE V–40 IN-FLIGHT 1/3 OCTAVE BAND SOUND PRESSURE LEVELS

After the payload and fourth stage have been balanced as a unit, it is recommended that the payload not be separated from the motor again. If it is impractical to keep the payload with the vehicle until launch, three means of detaching the payload from the fourth stage have been provided as follows:

a. The payload and adapter section may be removed from the forward end of the fourth stage motor.
b. The payload may be removed from the adapter section at the separation plane.
c. The payload may be removed from the adapter section forward of the separation plane.

4.1.12.2 Spin Rate

Spin forces must be considered in the design of the payload and its components. Three spin motors of different thrust levels are available that may be combined in five different configurations. Each configuration uses four motors to provide adequate spin stabilization. The spin rate of the fourth stage versus spin roll moment of inertia is presented in Figure V–41 for the five spin motor configurations. The inertia data for spin-up may be obtained from Tabel V–2 for the Scout fourth stage configuration. Figure V–42 presents angular acceleration at spin-up versus spin roll moment for the five spin motor configurations.

The minimum spin rate to be tolerated is determined by the accuracy required of the mission and will be a topic for discussion by the Mission Working Group.

4.1.12.3 Despin Systems

Should the Payload Agency desire the payload to be despun prior to separation, the despin mechanism must be designed so that the despin torque applied to the fourth stage motor case does not exceed 3350 inch-pounds. This torque is based on the torsional structural capability of the fourth stage motor case at elevated temperatures.

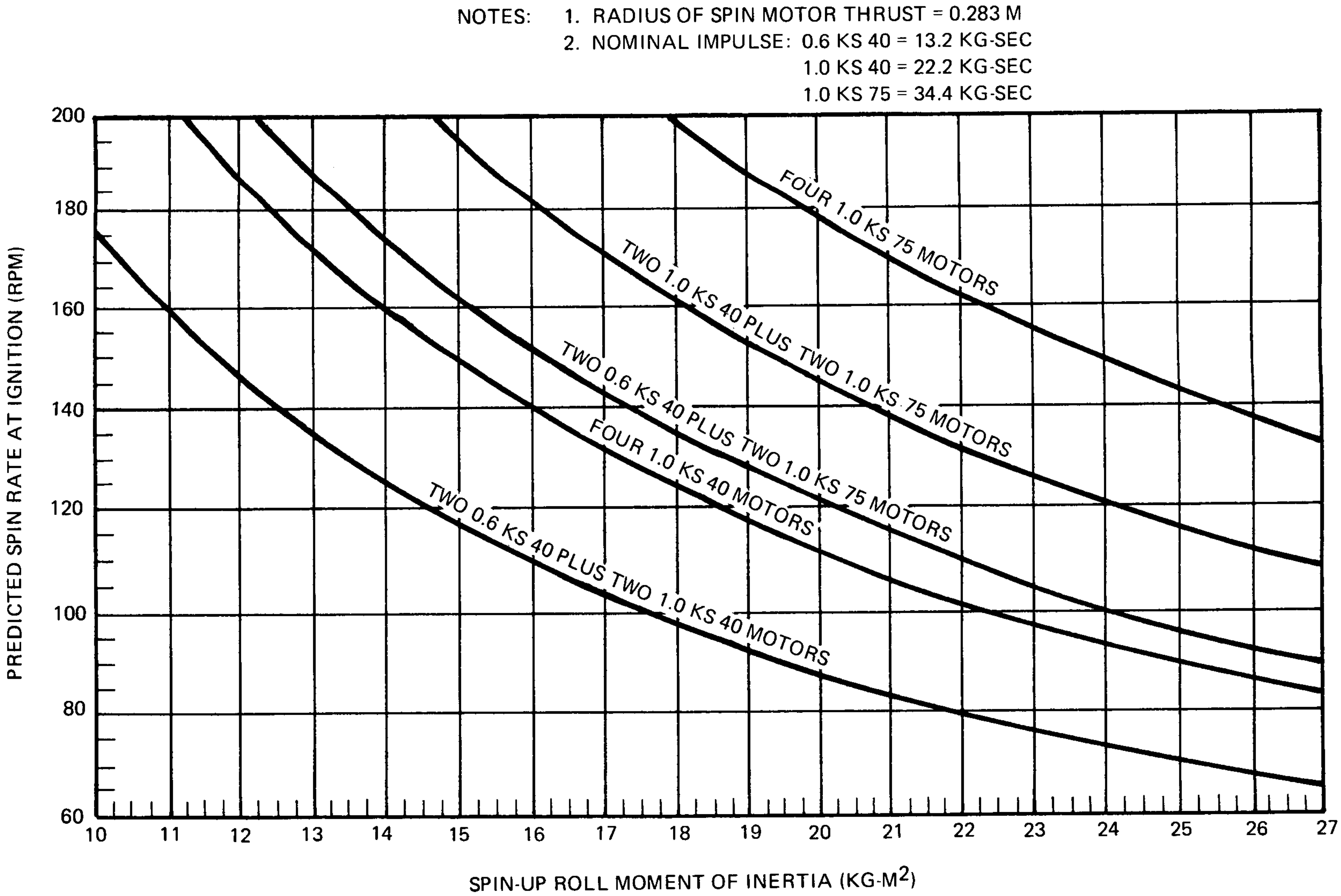

FIGURE V–41 PREDICTED SPIN RATE AT FOURTH STAGE IGNITION VS SPIN-UP ROLL MOMENT INERTIA

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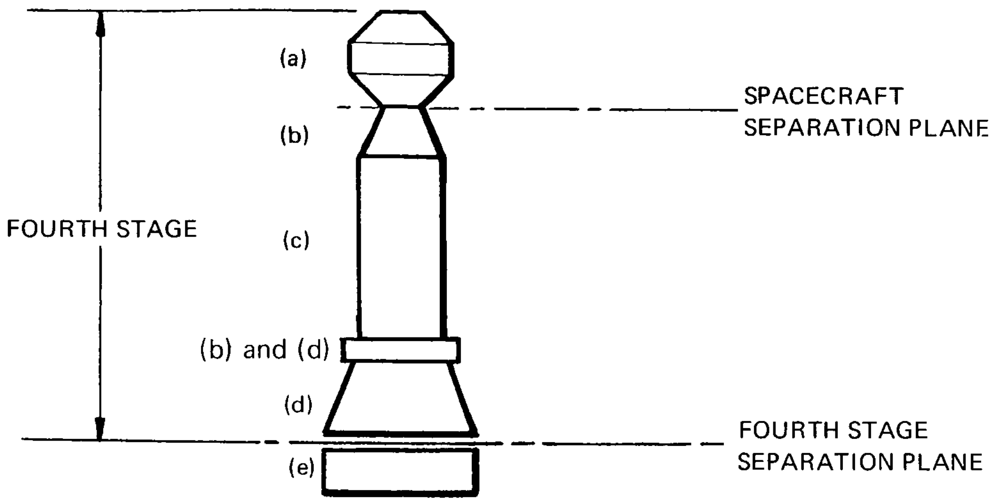
TABLE V-2 FOURTH STAGE WEIGHT AND INERTIA SUMMARY



ITEM	COMPONENT	WEIGHT (KILOGRAMS)	ROLL INERTIA (KG-M^2)	PITCH INERTIA (KG-M^2)	C.G. (SCOUT STA. –IN.)
(a)	SPACECRAFT				
(b)	ADAPTER AND ELECTRICAL SEPARATION SYSTEM COMPONENTS (REFER TO WEIGHT SUMMARY TABLE FOR APPLICABLE WEIGHT)				
(c)	FOURTH STAGE MOTOR				
	(1) ALTAIR (LOADED)	301.0	9.66	30.66	66.1
	(2) ALTAIR (EXPENDED)	25.1	1.17	5.59	69.9
(d)	UPPER "D" SECTION, FOURTH STAGE MODULE AND HARNESSES (LESS SPIN TABLE)	13.2	0.92	1.54	84.7
(e)	*SPIN TABLE	17.3	1.36	0.71	101.7

$(a) + (b) + (c_1) + (d) + (e)$ = SPIN-UP WEIGHT OR ROLL MOMENT OF INERTIA

$(a) + (b) + (c_2) + (d)$ = TOTAL SPIN MASS AFTER FOURTH STAGE SEPARATION AND BURNOUT

*SPIN-UP PORTION ONLY

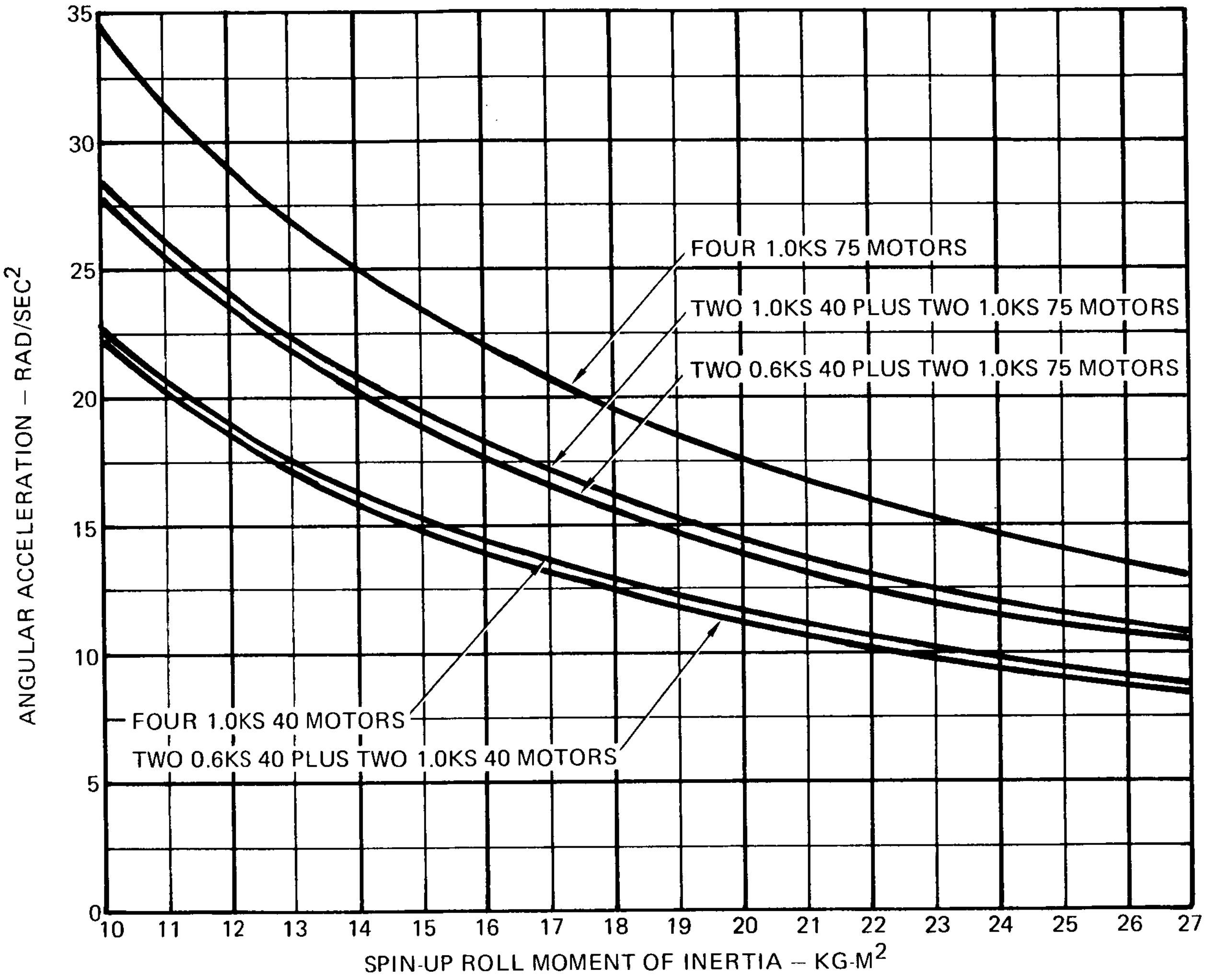


FIGURE V–42 NOMINAL ANGULAR ACCELERATION AT SPIN-UP VS SPIN-UP ROLL MOMENT OF INERTIA

4.1.13 ATTACHMENT TO MOTOR CASES

It may be necessary for the payload to use the fourth stage motor as a mounting base to attach some unique payload items: i.e., cradle supports for solar boom, special antenna rests, etc. Since the operation of the fourth stage and the motor case can be materially affected by these attachments, prior approval must be obtained through the Mission Work Group before such attachments are to be considered.

4.1.13.1 Adhesive Bonding

The following list of room temperature curing adhesives, which have been used on previous Scout flights, has been compiled to aid the Payload Agency in selecting a bonding agent that shall be used to attach a payload item to the motor case:

Make	Type
EA934 Dexter Corp. Pittsburg, Calif.	Epoxy Adhesive with 500^{o}F temperature service
RTV 88 General Electric Co. Waterford, N. Y.	Silicone rubber compound with limited strength up to 500^{o}F

4.1.13.2 Mechanical Attachment

Mechanical attachment to the fourth stage motor case may be made providing no motor modification is required. Mechanical attachment must be such that it provides for expansion of the motor case during the burning phase.

An example of this type attachment and the allowable areas for bonding are shown in Figure V–43.

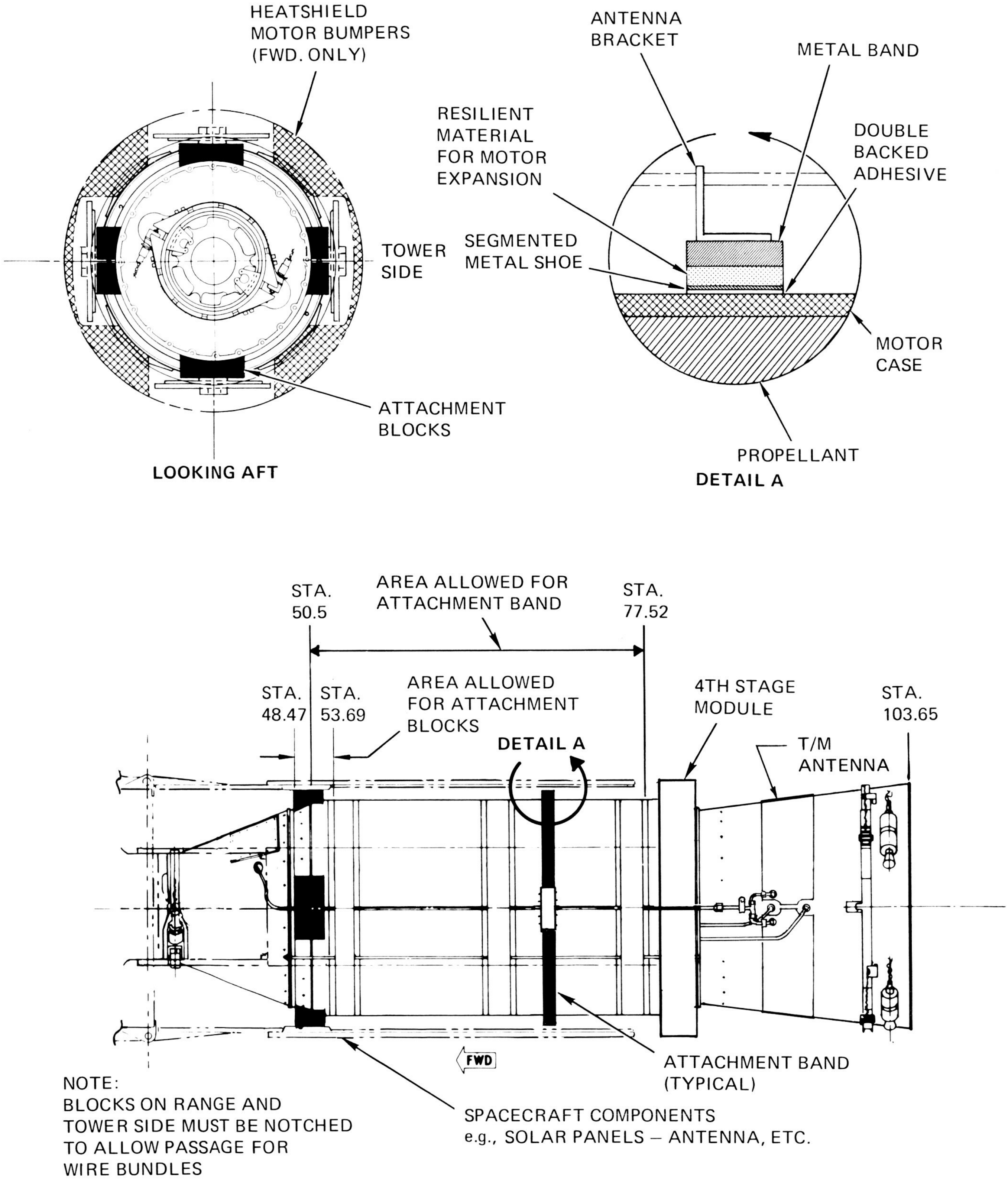


FIGURE V–43 ATTACHMENTS TO FOURTH STAGE MOTOR

200

4.1.14 WEIGHT BREAKDOWN SUMMARY

A weight breakdown summary of the Scout standard adapter separation systems and the separation system electrical components is presented in Table V–3.

TABLE V–3 WEIGHT BREAKDOWN SUMMARY

ITEM	WEIGHT (KILOGRAMS)	ROLL INERTIA ($KG\text{-}M^2$)	PITCH INERTIA ($KG\text{-}M^2$)	C.G. (SCOUT STA.)
"EG" SECTION ADAPTER (INCLUDING PAYLOAD SUPPORT RING)	2.4	0.06	0.05	41.0
SERIES 200 "E" SECTION ADAPTER (INCLUDING PAYLOAD SUPPORT RING)	4.93	0.10	0.08	40.4
SERIES 25 "E" SECTION ADAPTER (INCLUDING PAYLOAD SUPPORT RING)	6.1	0.48	0.28	41.3
SEPARATION SYSTEM ELECTRICAL COMPONENTS	3.3	0.27	0.16	82.2

5.0 SPACECRAFT TESTING

The tests presented in the following paragraphs are recommended mechanical tests and test levels to ensure that the spacecraft will survive the environment induced by the vehicle during boost. The recommended tests are based on experience of many launches and represent the tests used most by other agencies. Other tests could be defined which fully satisfy these environmental requirements. The actual tests and test levels to fully qualify the spacecraft to be launched on the Scout vehicle are the responsibility of the Spacecraft Agency.

For those programs which have an engineering model or prototype spacecraft, it is recommended that it be used in all qualification level testing. It is recommended that flight hardware be tested to flight acceptance test levels. Should the program have only a protoflight spacecraft, it is left up to the discretion of the spacecraft agency whether or not to test flight hardware to qualification levels of testing.

The engineering model or prototype spacecraft should be subjected to Scout random and sine vibration testing to qualification levels in all three axes. The flight or protoflight units should be subjected to Scout flight acceptance test random and sine vibration levels appropriately "notched" for sine.

5.1 ACCELERATION TESTING

The spacecraft shall be operating during the acceleration tests in each of three axes. Acceleration applied along the thrust axis should be 1-1/2 times the maximum calculated level (Figure V—33 or V—34), for the spacecraft weight and center of gravity (CG), for a duration of three minutes. Acceleration applied in each of the lateral axes should be at 1.5 g for a duration of one minute. The acceleration gradient from the CG should not exceed plus or minus ten percent.

5.2 SHOCK TESTING

The spacecraft shall be operating during exposure to shock tests in the thrust axis. The spacecraft shall be subjected to three terminal sawtooth pulses of six milliseconds duration applied along the thrust axis. The amplitude is a function of spacecraft weight and is indicated by the Design Qualification Test Level curve of Figure V—44. The terminal sawtooth pulse required to achieve the observed shock amplitude is presented in Figure V—44. Careful attention should be given to ensure that the shock pulse is applied in one direction only. A reversal in direction of the applied loading may result in response loads greater than the capability of the structure. It is not required that the flight spacecraft be subjected to mechanical shock tests.

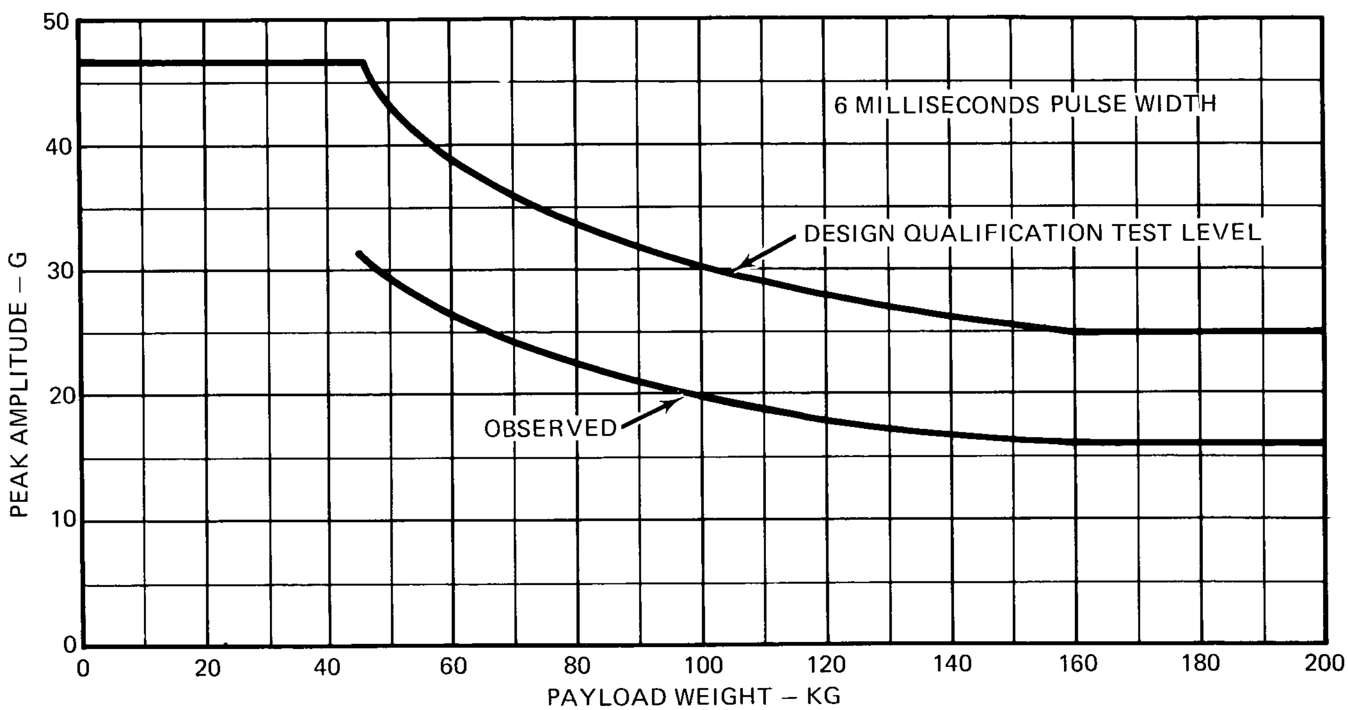


FIGURE V–44 TERMINAL SAWTOOTH INPUT REQUIRED TO ACHIEVE SHOCK AMPLITUDE FOR TRANSITION SECTION – LONGITUDINAL AXIS

5.3 VIBRATION TESTING

The spacecraft (including the adapter/separation assembly structure which attaches the payload to the fourth-stage motor) should be operating during exposure to the vibration test. The test levels apply at the interface of the forward shoulder of the fourth stage motor. The tests should be performed with a structurally and dynamically similar adapter/separation assembly installed.

5.3.1 Sinusoidal Vibration Testing

The sinusoidal vibration test shall be conducted by sweeping at a logarithmic rate from the lowest to the highest frequency once for each range specified (see Figures V—45 and V—46).

All spacecraft should be critically reviewed prior to sinusoidal vibration testing to preclude failure from unrealistic loads experienced during testing. Since a spacecraft influences its own vibrational environment and because of the artificial (non-flight) constraints placed on the spacecraft during testing, excessive loads may be experienced at resonant frequencies. Figures V—45 and 46 do not take into account possible reduction at resonance because the effect is a function of the dynamic characteristics of each spacecraft. In order to predict the magnitude of the vehicle flight loads, a dynamic response analysis is conducted by either Vought Corporation or the Spacecraft Agency or manufacturer. The spacecraft mathematical model required to perform such a response analysis can be supplied in terms of either the basic mass and stiffness properties of the spacecraft or as modal data of the spacecraft. The spacecraft mathematical model is coupled with that of other components of the Scout vehicle and a response study is conducted to determine maximum expected flight loads at the forward end of the fourth stage motor. Where it can be shown by such a dynamic response analysis that a particular spacecraft in combination with the launch vehicle experiences loads at all critical flight conditions less severe than those induced by the specified vibration test, the test spectrum may be "notched" in the frequency bands of resonances of primary structure. The bandwidth of the notch should be determined immediately before the test by a low-level sine sweep survey. The survey level should be as high as possible to minimize non-linear effects. The notch bandwidth must be narrow enough to develop 90% of the allowable loads at the band edges.

5.3.1.1 Qualification Test

Apply one sweep in each of three axes at a logarithmic sweep rate not greater than two octaves per minute.

5.3.1.2 Flight Acceptance Test

Apply one sweep in each of three axes at a logarithmic sweep rate not greater than four octaves per minute.

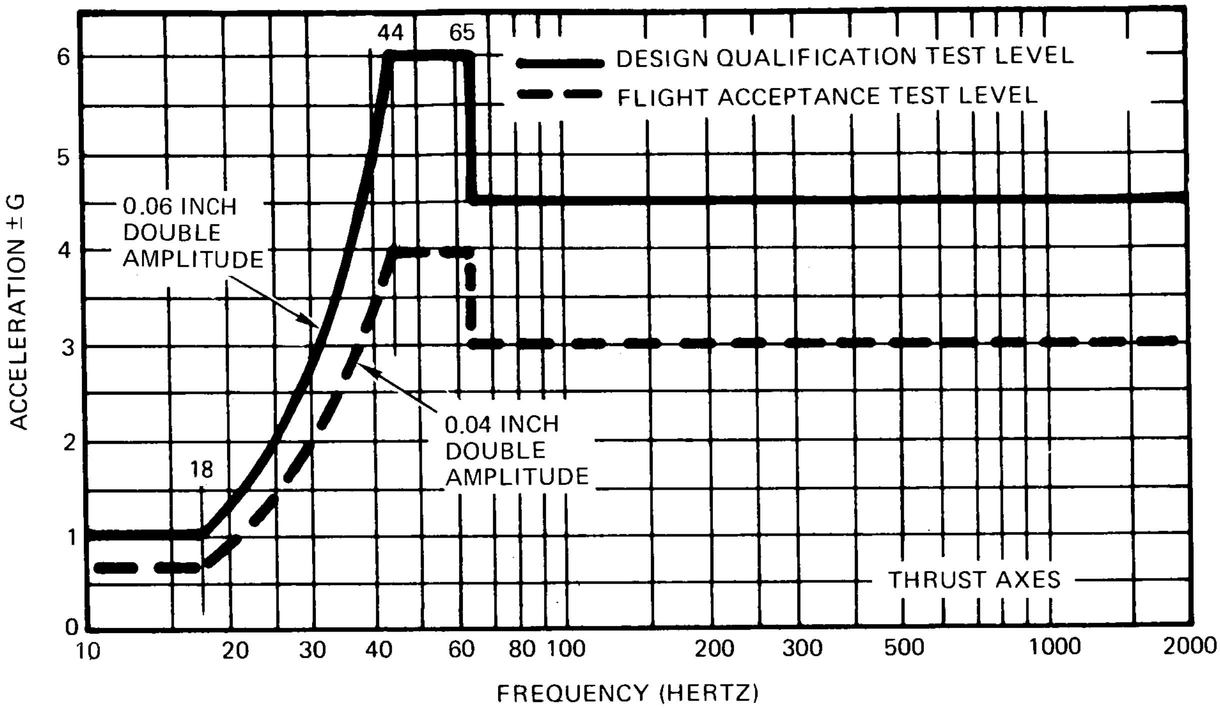


FIGURE V–45 SINUSOIDAL VIBRATION TEST LEVELS – THRUST AXIS

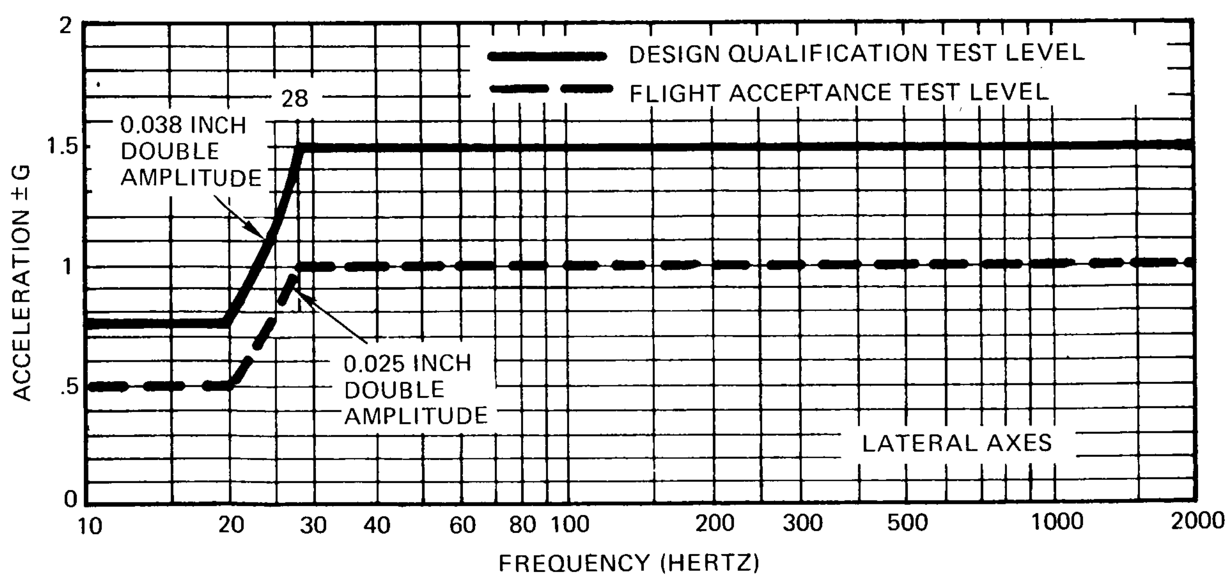


FIGURE V–46 SINUSOIDAL VIBRATION TEST LEVELS – LATERAL AXIS

5.3.2 Random Vibration Testing

Gaussian random vibration shall be applied as specified in Figure V–47.

5.3.2.1 Qualification Test

Apply gaussian random in each of three axes for two minutes.

5.3.2.2 Flight Acceptance Test

Apply gaussian random in each of three axes for one minute.

5.3.3 Acoustic Noise Testing

Smaller spacecraft, typical of Scout payloads, are more likely to be sensitive to mechanically transmitted random vibration than acoustic noise. However, small spacecraft equipped with large area, low surface mass density components and delicate membranes or diaphrams should be considered for acoustic testing. For Scout spacecraft, the acoustic test may not be used as a substitute for the random vibration test.

5.3.3.1 Qualification Test

Acoustic noise test levels as specified in Table V–4 for two minutesduration.

5.3.3.2 Flight Acceptance Test

Acoustic noise test levels as specified in Table V–5 for one minute duration.

5.4 SPIN

The spacecraft should be subjected to 1.5 times the nominal flight spin rate. During dynamic balancing of the spacecraft/fourth stage assembly, in the field, the spacecraft will be subjected to a spin rate equal to the flight three sigma value as a minimum.

5.5 SPACECRAFT AND VEHICLE COMPATIBILITY TEST

To ensure a reliable separation of the heatshield from the vehicle, with no damage to either the spacecraft or the vehicle, a heatshield fit test is conducted on each spacecraft configuration. In some extreme cases, where the complexity of the spacecraft warrants such action, the Mission Working Group may recommend a heatshield ejection test be performed.

These tests are conducted at the Vought Corporation facility in Dallas, Texas and consume a period of approximately two calender days. During both the fit test and the ejection test, it is essential that the envelope of the spacecraft, including the correct position of any antenna, wiring bundles, solar panels, etc., be the same as the actual spacecraft. However, the use of the flight spacecraft should be avoided to prevent possible handling damage.

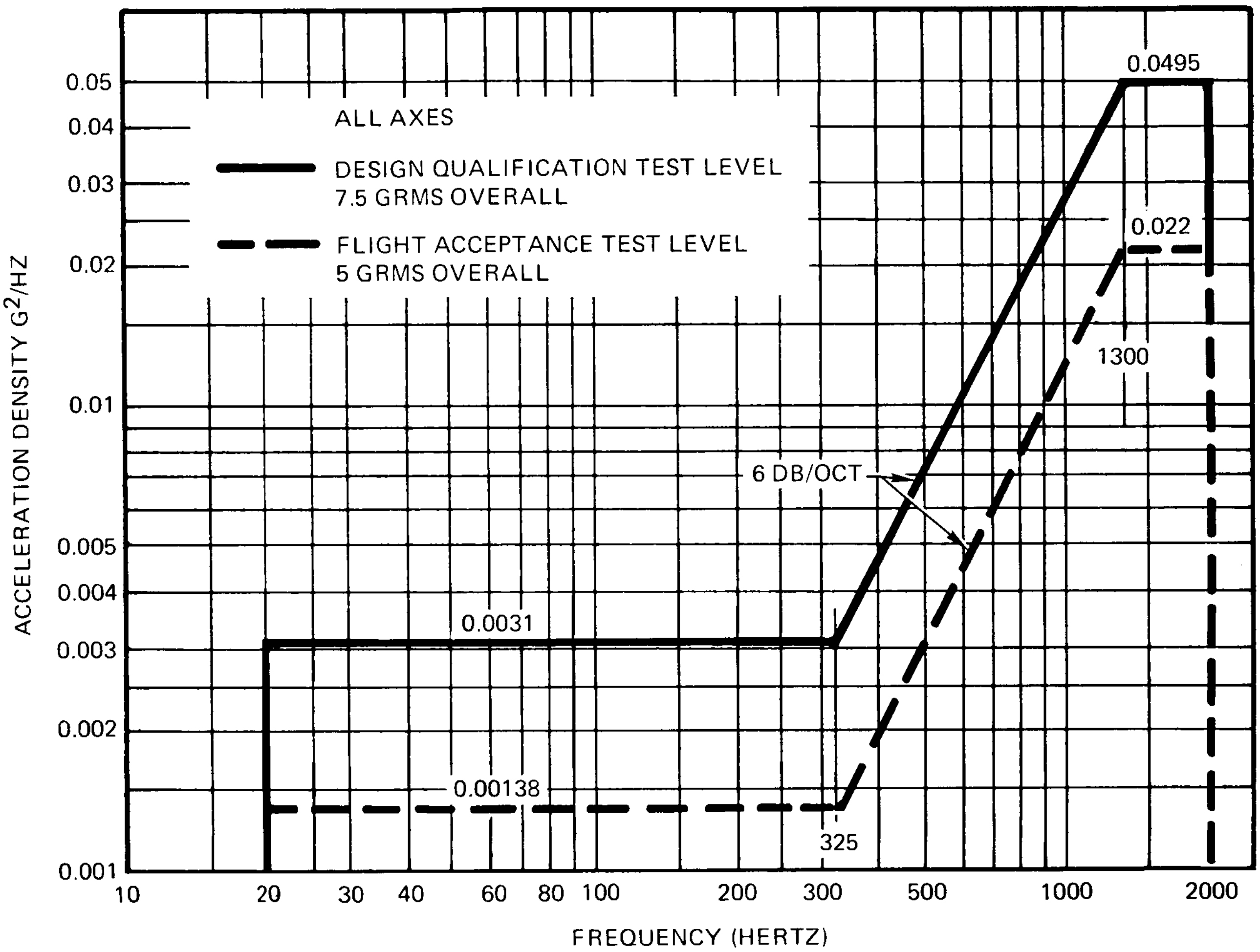


FIGURE V–47 RANDOM VIBRATION TEST LEVELS – ALL AXES

TABLE V-4 SPACECRAFT QUALIFICATION—ACOUSTIC NOISE

OCTAVE BAND CENTER FREQUENCY (Hz)	SOUND PRESSURE LEVEL (dB, REF. 20μ N/m^2)		DURATION (MINUTES)
	EXTERIOR OF SHROUD	INTERIOR OF SHROUD	
31.5	140	127	2
63.0	141	127	
125.0	146	129	
250.0	150	134	
500.0	149	132	
1000.0	147	132	
2000.0	145	132	
4000.0	144	132	
8000.0	144	132	
OVERALL	156	141	

TABLE V—5 SPACECRAFT FLIGHT ACCEPTANCE—ACOUSTIC NOISE

OCTAVE BAND CENTER FREQUENCY (Hz)	SOUND PRESSURE LEVEL (dB, REF. 20μ N/m^2)		DURATION (MINUTES)
	EXTERIOR OF SHROUD	INTERIOR OF SHROUD	
31.5	136	123	1
63.0	137	123	
125.0	142	125	
250.0	146	130	
500.0	145	128	
1000.0	143	128	
2000.0	141	128	
4000.0	140	128	
8000.0	140	128	
OVERALL	152	137	

5.5.1 Fit Test

The spacecraft and spacecraft adapter are assembled in a vertical position to the Scout transition "D" section, a fourth stage motor case, and when required, an "F" section and a fifth stage motor case. A fit test is to ensure that adequate clearances exist between the heatshield and spacecraft, motor cases and any wiring bundles associated with the Scout ignition system or spacecraft. Also, the position of the spacecraft with the heatshield in position is checked during the fit test.

5.5.2 Ejection Test

In cases of complex spacecraft configurations, the Mission Working Group may require a heatshield ejection test be performed after the fit test to demonstrate that the heatshield will separate with no interferences to the spacecraft. This ejection test is performed on a simulated "zero g" fixture. The heatshield, attached to "D" section and enclosing the final stage(s) and spacecraft, is allowed to fall free while it separates. The vehicle/platform assembly slides freely down vertical rails until a braking system is automatically engaged which brings the vehicle/platform assembly to a smooth stop, with no abrupt deceleration shock, with a maximum deceleration of 6g. The relative motion between the heatshield and final stage(s) is then recorded. All heatshield hardware used in the ejection tests is flight hardware, and a safety net is arranged to catch the heatshield halves upon separation. High-speed motion pictures are made of the separation, with minimum camera coverage being one camera looking aft along the vehicle centerline from above, and one camera focused on the horizontal separation plane at the approximate height of the CG of the heatshield. Additional cameras may be used as required during the ejection test to view critical areas of spacecraft, such as antenna. A still photograph is also made of theseparation.

6.0 ELECTRICAL DESIGN CRITERIA

This section identifies the payload/Ground Equipment electrical interface and the payload design radio frequency interference (RFI) criteria.

6.1 PAYLOAD UMBILICAL CONNECTOR

The connector used on the umbilical, which attaches to the payload is a Bendix "twist/pull" pigmy electrical connector that requires a pull load dependent on the size of the connector and direction of the pull. This type of connector has been specifically qualified for use on Scout. Figure V–48 lists the shell and arrangement of pins used by the umbilical. The payload half (female) of this connector must be provided by the Payload Agency. The Payload Agency must define, through the Mission Working Group, the Payload to Blockhouse wiring requirements at least six months prior to launch.

6.1.1 Payload Umbilical Connector Bracket Design

The payload umbilical connector mounting bracket design must allow the following retraction forces:

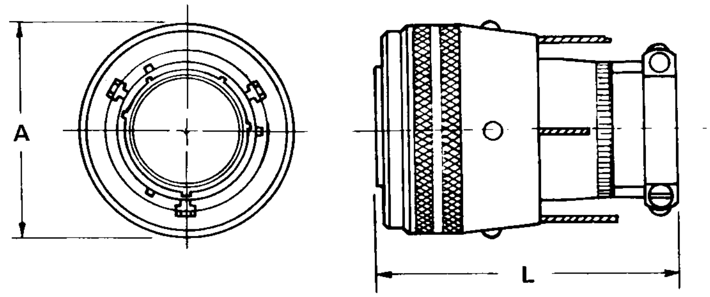
Shell Size	Static Retraction Force
12	5.4 Kilograms (12 Pounds)
16	11.3 Kilograms (25 Pounds)
24	22.7 Kilograms (50 Pounds)

Mounting bracket and payload connector attaching hardware must adhere to the maximum dimensions shown in Figure V–49 to assure proper operation of the payload umbilical connector.

6.2 PAYLOAD DESIGN RFI CRITERIA

Radio frequency interference (RFI) compatibility is required between the payload and launch vehicle. The vehicle RF systems characteristics are provided in Table V–6. The command destruct receiver and the beacon are required to operate until third stage separation occurs. It should be noted from the table that the sensitivity levels of the receivers are such that conformance to MIL-I-6181 or MIL-I-26600 by the transmitters with respect to spurious emanations is not necessarily sufficient to prevent interference should one of the spurious signals coincide with a receiver frequency. The relative amplitudes of the "D" section telemetry transmitter carrier and spurious emanations are shown in Figure V–50. Utilization of these frequencies by the payload receivers should be avoided. They should also avoid frequencies within ±75 megahertz of the radar beacon transmitter carrier and fifth stage telesponder carrier due to the wide band characteristics of the beacon pulse transmissions.

BENDIX TWIST/PULL TYPE CONNECTOR NO. 72-49070_ OR NO. 72-3395–

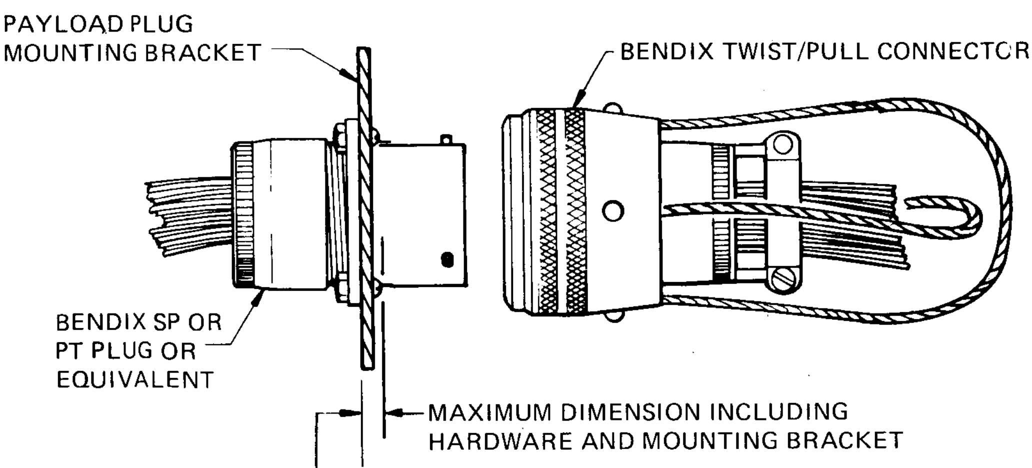


NOTE: BENDIX TWIST/PULL TYPE CONNECTOR IS PART OF THE VEHICLE FURNISHED "3" FT. PAYLOAD UMBILICAL CABLE

SHELL SIZE	BASE NO.	A DIAMETER	L MAX.
10	72-3395	1.125	2.625
12	72-490703	1.281	2.625
14	72-3395	1.438	2.625
16	72-490705	1.531	2.817
18	72-490706	1.672	2.817
20	72-490707	1.781	2.956
22	72-3395	1.875	3.076
24	72-3395	2.134	3.076

INSERT ARRANGEMENT	TOTAL CONTACTS	CONTACT SIZE	
		20	16
10-6	6	6	
12-8	8	8	
12-10	10	10	
14-18	18	18	
16-23	23	22	1
16-26	26	26	
18-30	30	29	1
18-32	32	32	
20-16	16		16
20-24	24	24	
20-39	39	37	2
20-41	41	41	
22-36	36	36	
24-61	61	61	

FIGURE V–48 BENDIX TWIST/PULL UMBILICAL CONNECTORS



PLUG SHELL SIZE	SP DIMENSION	PT DIMENSION
10 THRU 18	0.125 INCH	0.094 INCH
20 THRU 24	0.219 INCH	0.219 INCH

FIGURE V–49 PAYLOAD UMBILICAL PLUG BRACKET DIMENSIONS

TABLE V–6 VEHICLE RF SYSTEMS

RADAR BEACON TRANSMITTER		
CENTER FREQUENCY	5555.0 ± 4.0 MHz	
MODULATION	AMPLITUDE, SINGLE PULSE	
MAXIMUM POWER OUTPUT	1000 WATTS PEAK	
MINIMUM POWER OUTPUT	400 WATTS PEAK	
ANTENNA GAIN TOWARD PAYLOAD WITH RESPECT TO ISOTROPIC RADIATOR	0 DB	
RADAR BEACON RECEIVER		
CENTER FREQUENCY	5480.0 ± 3 MHz	
IMAGE FREQUENCY	5586.0 MHz	
LOCAL OSCILLATOR FREQUENCY	5533 MHz	
MAXIMUM RF BANDWIDTH		
3 DB POINTS	15 MHz	
30 DB POINTS	40 MHz	
NOMINAL SENSITIVITY	–68 DBM	
ANTENNA GAIN TOWARD PAYLOAD WITH RESPECT TO AN ISOTROPIC RADIATOR	0 DB	
COMMAND DESTRUCT RECEIVER		
CENTER FREQUENCY		
WFC LAUNCH	412.0 MHz ± 0.01%	
VAFB LAUNCH	416.5 MHz ± 0.01%	
IMAGE FREQUENCY (SENSITIVITY –40 DBM)		
WFC LAUNCH	268 MHz	
VAFB LAUNCH	272 MHz	
LOCAL OSCILLATOR FREQUENCY		
WFC LAUNCH	340 MHz	
VAFB LAUNCH	340 MHz	
MAXIMUM RF BANDWIDTH	6 DB POINTS	60 DB POINTS
WFC LAUNCH	650 KHz	2.8 MHz
VAFB LAUNCH	200 KHz	360 KHz
NOMINAL SENSITIVITY		
WFC LAUNCH	–101 DBM	
VAFB LAUNCH	–107 DBM	
ANTENNA GAIN TOWARD PAYLOAD WITH RESPECT TO AN ISOTROPIC RADIATOR	0 DB	

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TABLE V–6 VEHICLE RF SYSTEMS (CONTINUED)

"D" SECTION TELEMETRY TRANSMITTER	
CENTER FREQUENCY	2230.5 MHZ ± 0.002%
MAXIMUM RF BANDWIDTH	500 KHZ
MODULATION	PAM/FM/FM
MAXIMUM POWER OUTPUT	7 WATTS
MINIMUM POWER OUTPUT	5 WATTS
ANTENNA GAIN TOWARD PAYLOAD WITH RESPECT TO ISOTROPIC RADIATOR	0 DB
FOURTH STAGE TELEMETRY TRANSMITTER	
CENTER FREQUENCY	2210.5 MHZ
MAXIMUM RF BANDWIDTH	500 KHZ
MODULATION	FM/FM
MAXIMUM POWER OUTPUT	7.0 WATTS
MINIMUM POWER OUTPUT	5.0 WATTS
ANTENNA GAIN TOWARD PAYLOAD WITH RESPECT TO ISOTROPIC RADIATOR	0 DB
FIFTH STAGE TELESPONDER TRANSMITTER	
CENTER FREQUENCY	5690 ± 4.0 MHZ
MODULATION	AMPLITUDE, DOUBLE PULSE
MAXIMUM POWER OUTPUT	1000 WATTS PEAK
MINIMUM POWER OUTPUT	400 WATTS PEAK
ANTENNA GAIN TOWARD PAYLOAD WITH RESPECT TO ISOTROPIC RADIATOR	–10 DB (ESTIMATED)
FIFTH STAGE TELESPONDER RECEIVER	
CENTER FREQUENCY	5825.0 ± 3 MHZ
IMAGE FREQUENCY	5635.0 MHZ
LOCAL OSCILLATOR FREQUENCY	5580.0 MHZ
MAXIMUM RF FREQUENCY	
3 DB POINTS	15 MHZ
30 DB POINTS	40 MHZ
NOMINAL SENSITIVITY	–68 DBM
ANTENNA GAIN TOWARD PAYLOAD WITH RESPECT TO AN ISOTROPIC RADIATOR	–10 DB (ESTIMATED)

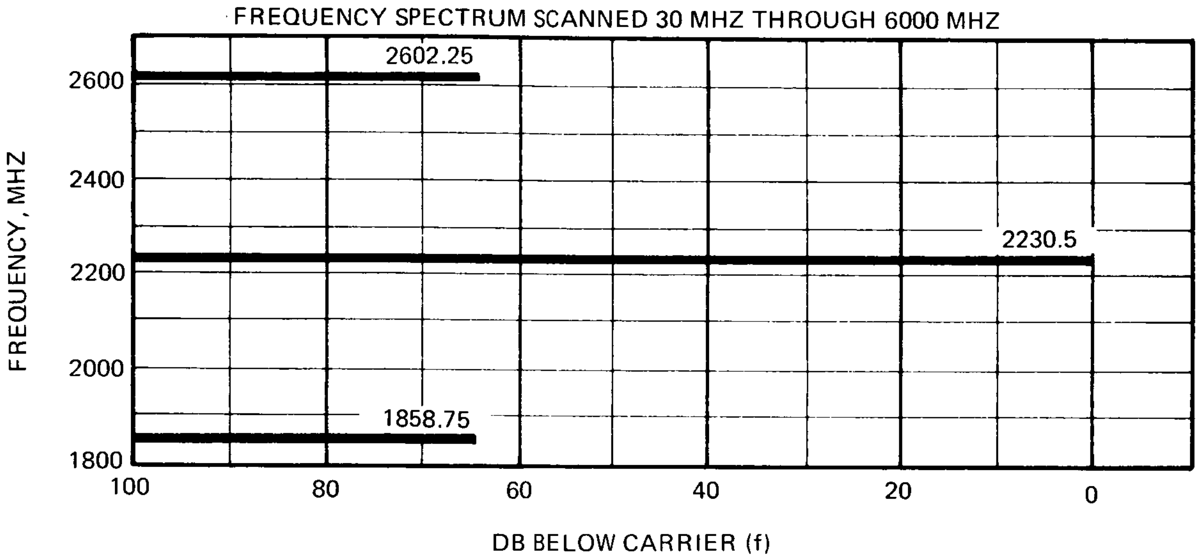


FIGURE V–50 RELATIVE AMPLITUDE OF ANTENNA CONDUCTED SIGNALS OF TYPICAL SCOUT TELEMETRY TRANSMITTER

6.2.1
RFI Tests

A preliminary RFI test is mandatory prior to final vehicle launch processing. This test is normally scheduled to occur during final vehicle systems check-out at Vought Corporation. If vehicle and payload schedules are incompatible, alternate arrangements may be made. Final RFI tests are conducted in the field during on-pad operation.

6.3
OPERATIONAL CONSTRAINTS

When a payload contains a timing sequencer that is independent of vehicle timing (first motion, sequencer operation, etc.) the payload timer must be capable of remote reset in the event of a hold in vehicle count.

If a hold in the countdown is required, it is frequently necessary to reset certain sequencers. For this reason, it is mandatory that all payload sequencers be of a type that are easily reset.

7.0 INTRODUCTION

NASA's decision to replace the present Spacecraft Tracking and Data Network (STDN) with the Tracking Data Relay Satellite System (TDRSS) will necessitate additions and changes to the communication systems of Scout launched satellites. These additions and changes will result in space and weight changes for the spacecraft.

The Scout program has conducted a study to identify a standard TDRSS communication module/separation system configuration that could be flown with Scout launched spacecraft. The module would contain the major elements (antenna and transponder) of the satellite communication system.

7.1 ANTENNA

Ball Aerospace Systems Division, under NASA contract, has studied two microprocessor controlled Electronic Switching Spherical Array (ESSA) antenna system configurations that could be operated in the acquisition/track mode on a spinning spacecraft. These antenna configurations are shown in Figure V–51.

7.1.1 r-f Coverage

The actual r-f look angle for a given spacecraft may be less than that shown in Figure V–51. This variation could result from spacecraft obstructions such as structure, solar panels, experiment booms, etc. It may also be expected that continuous coverage would not be possible as a result of these obstructions.

7.2 TRANSPONDER

The remaining major component of the satellite communication system is the NASA standard S-band TDRSS transponder, shown in Figure V–52. The transponder is available with r-f power outputs of 1.0, 2.5, and 5.0 watts. The input power for these units is 19, 28, and 37 watts, respectively, of which 12 watts is required to power the receiver.

7.3 CONFIG-URATIONS

Figures V–53 and V–54 identify two possible standard configurations for a Scout class TDRSS module and payload separation system.

Figure V–53 uses the existing Scout series 200 "E" section in combination with the toroidal ESSA antenna module. This configuration has the potential to provide more r-f coverage, higher antenna gain and the least intrusion into the payload envelope. The disadvantages of this configuration are that it is the heaviest and, in some cases, could offer restrictions on the viewing area of an experiment. The use of this configuration would require a 42 inch diameter heatshield.

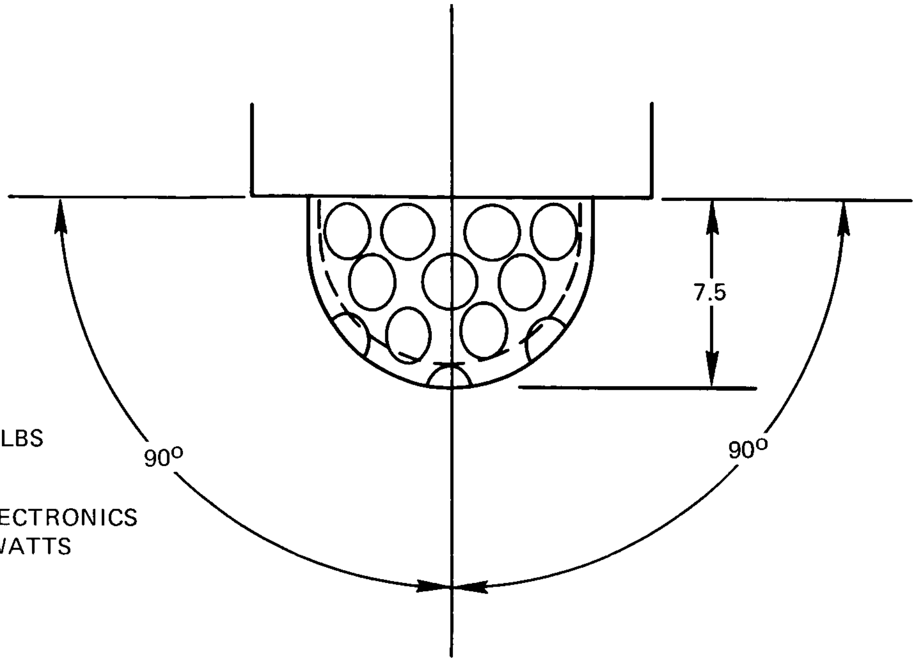
Figure V–54 uses a hemispherical ESSA antenna, 15 inches in diameter, in combination with a new separation system. While the separation clamp for this system would be the same as that of the existing Scout series 25 "E" section, the remaining structure is different. If the mission objectives are such that the reduced r-f coverage and system gain of this configuration can be tolerated, it offers an advantage in weight. This configuration could be used in either a 34 inch or 42 inch diameter heatshield.

A summary of the weights of these two systems is summarized in the following table:

	Toroidal	Hemispherical
Transponder	15.7	15.7
Antenna System	19.8	6.5
Module Structures	12.6	5.2
Separation System	9.7	11.2
Electrical Harness	3.0	3.0
Total	60.8 Pounds	41.6 Pounds

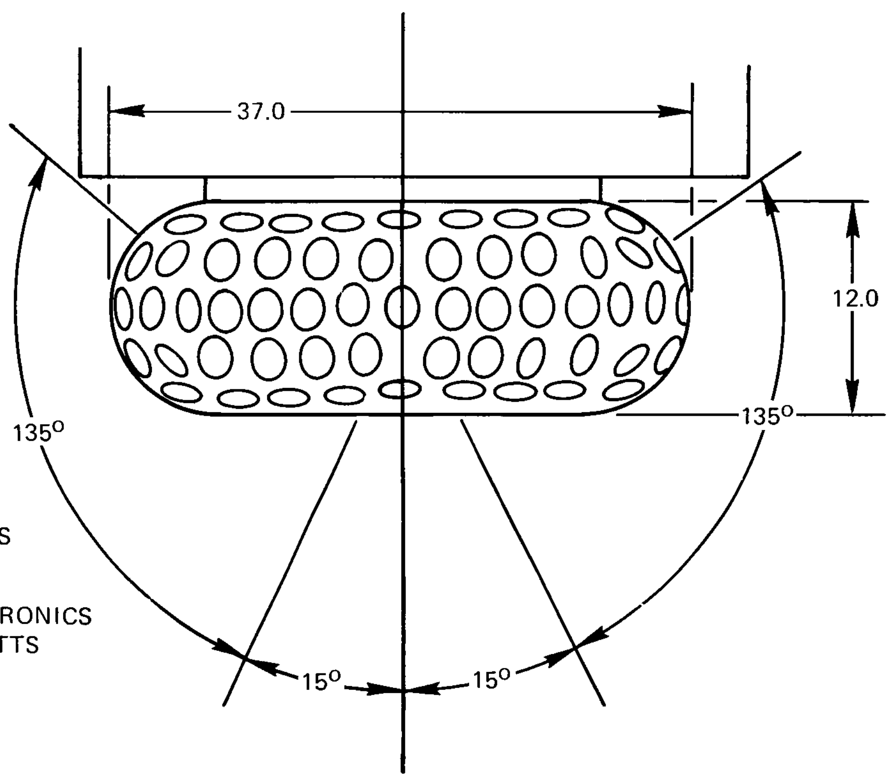
7.4 HEATSHIELDS

Two Scout heatshield configurations are presently available which would be compatible with a TDRSS module; the 42 inch diameter, –45 nose station heatshield and the 34 inch diameter, –40 nose station heatshield. A study has been completed which shows it is feasible to add 10 inches and 15 inches, respectively, to these heatshields. The nose of both heatshields would then be station –55. This increase in length would compensate for that required by the TDRSS module.



WEIGHT = 6.5 LBS
GAIN = 8.7 dBi
STEERING ELECTRONICS POWER = 7.4 WATTS

HEMISPHERICAL ANTENNA



WEIGHT = 19.8 LBS
GAIN = 14.5 dBi
STEERING ELECTRONICS POWER = 11.5 WATTS

TOROIDAL ANTENNA

FIGURE V–51 RADIATION COVERAGE

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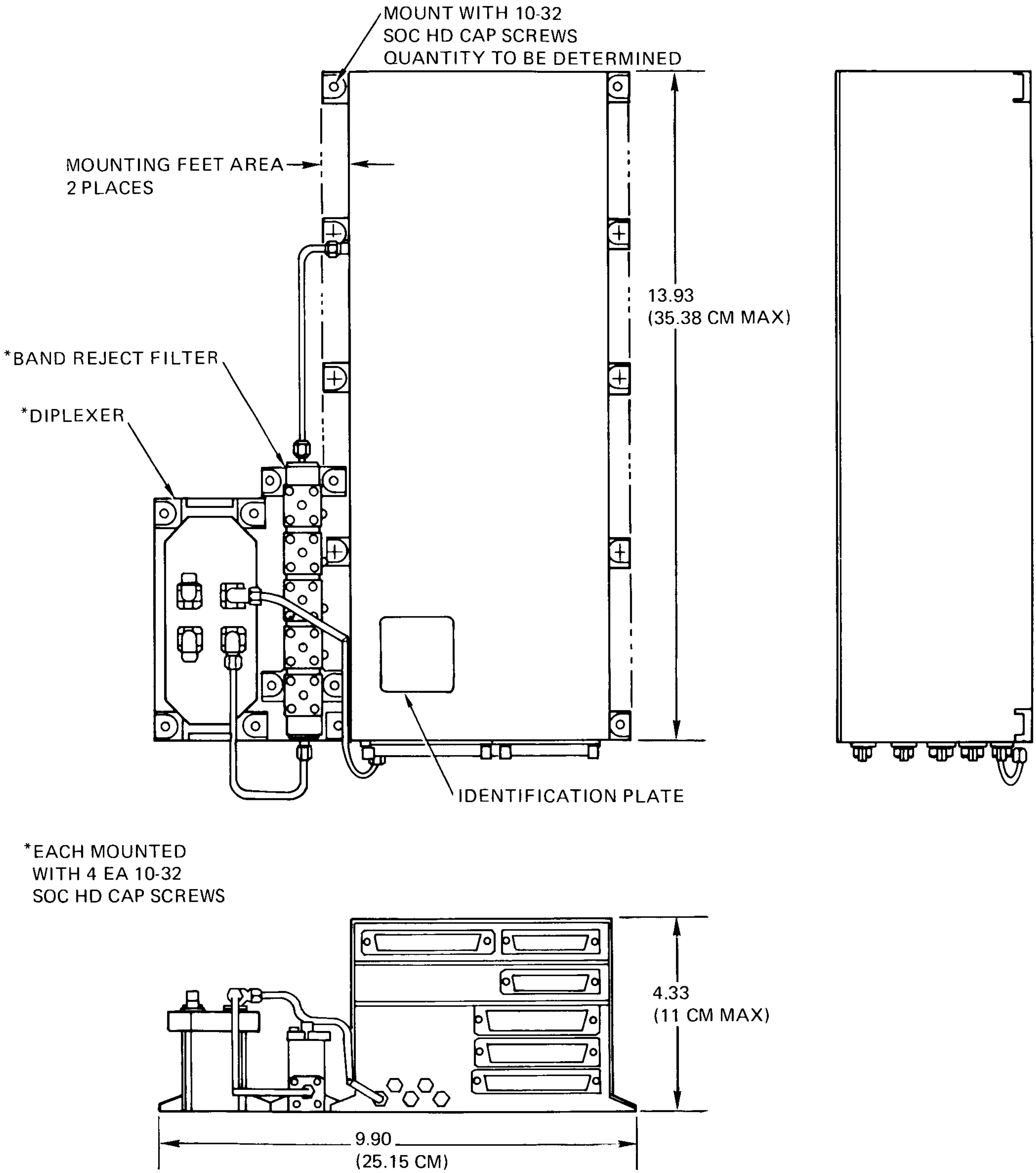

FIGURE V–52 TDRSS TRANSPONDER OUTLINE

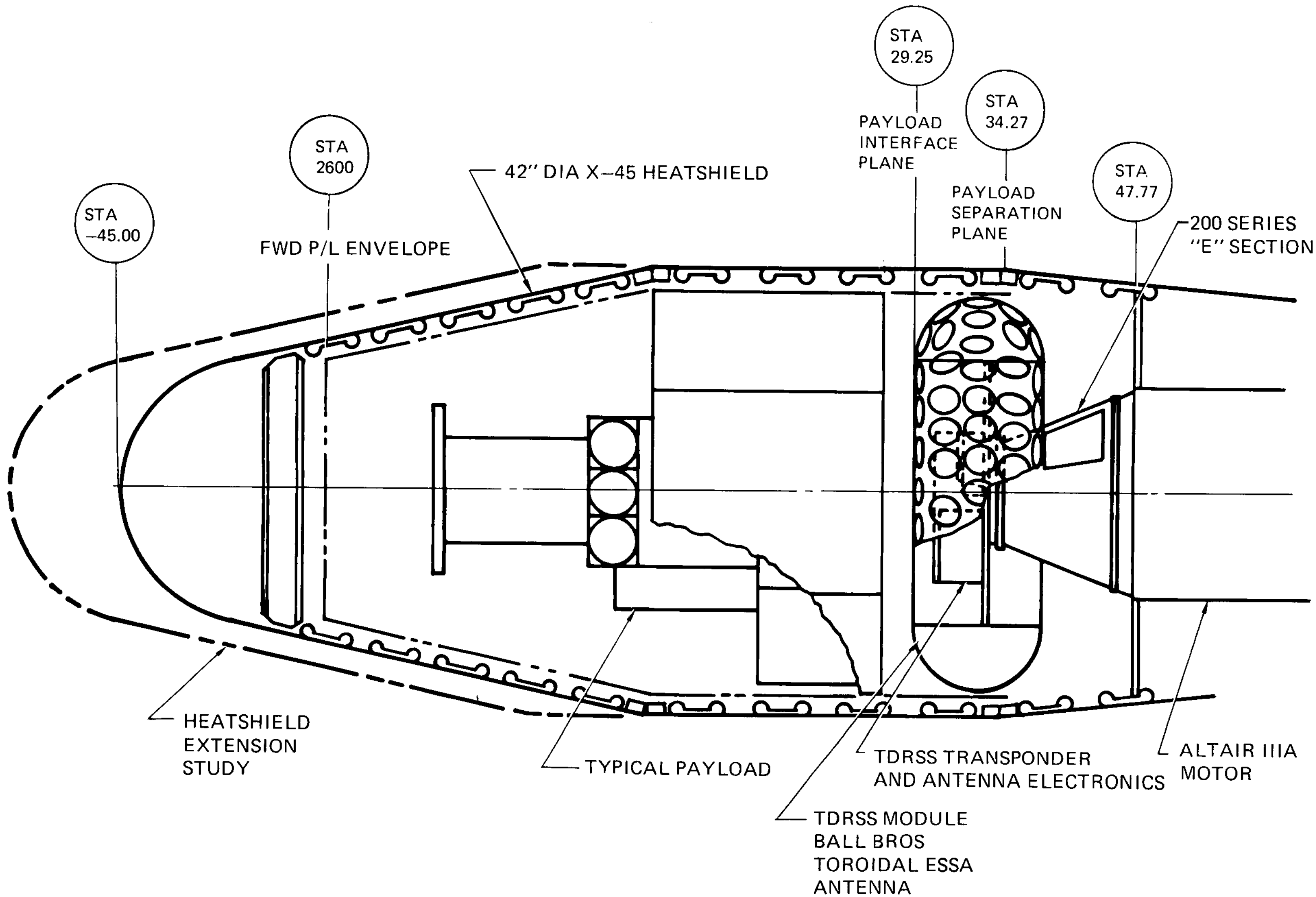

FIGURE V–53 PAYLOAD ENVELOPE WITH TOROIDAL ANTENNA TDRSS MODULE

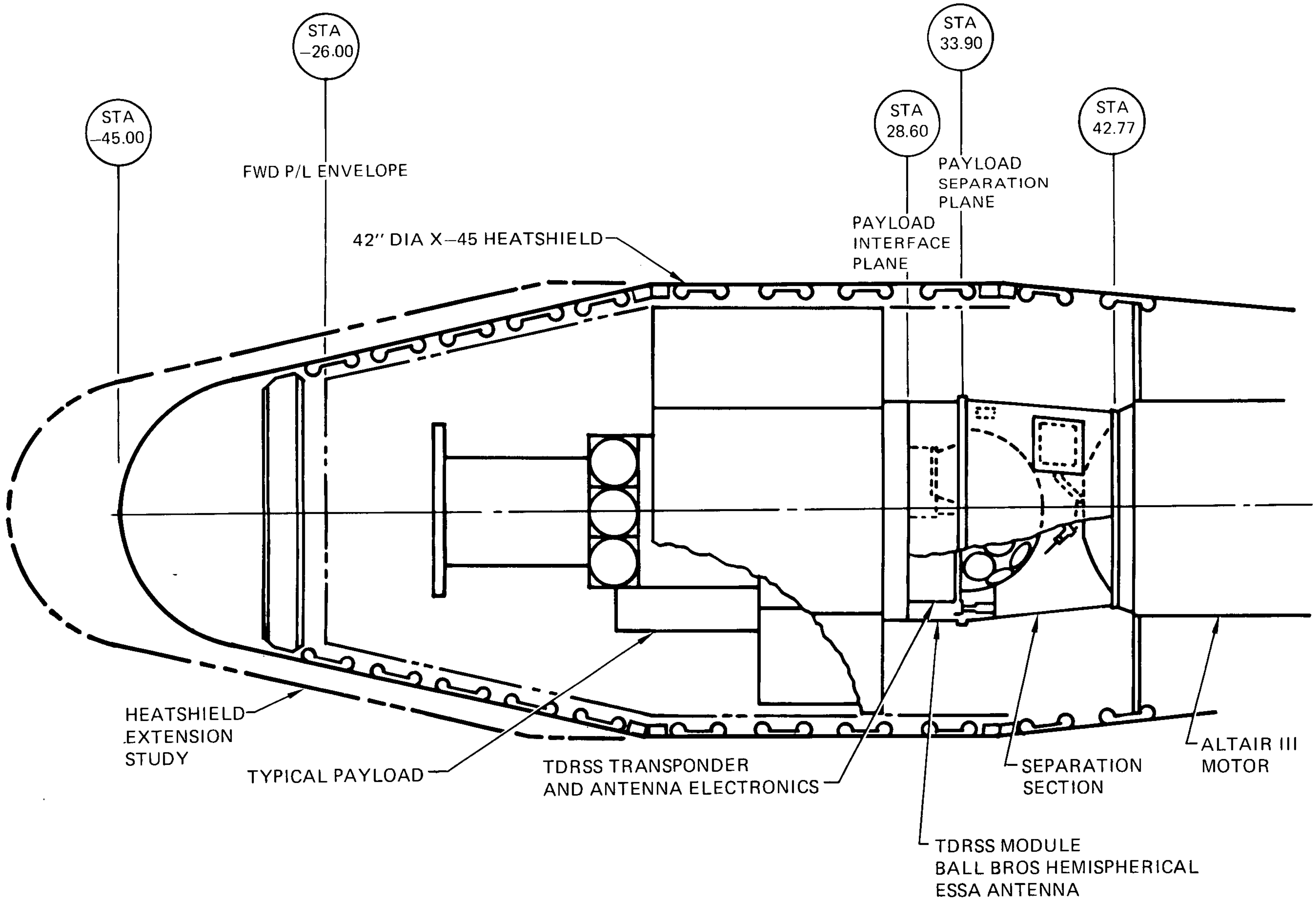

FIGURE V–54 PAYLOAD ENVELOPE WITH HEMISPHERICAL ANTENNA TDRSS MODULE